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RINGSAT SPACECRAFT
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1.0 INTRODUCTION

Radio-Frequency Interference (RFI) presents a serious problem to naval communications. Unauthorized transmissions on the same frequencies that FLTSAT or LEASAT use cause a loss of one or more transponders on a satellite. The problem is not correctable until the emitter is located and turned off, or until it turns off on its own. The present geolocation system cannot deal with the RFI problem adequately. This causes the Navy to lose operational capability and money. The proposed Radio Frequency Interference Navy Geolocation Satellite (RINGSAT) would locate the source of the interference within 3 hours, an outstanding improvement over current equipment. Furthermore, RINGSAT can be built quickly, at minimal cost, and can be concurrently used for other military missions.

Currently, geolocation to identify the RFI is done at Naval Space Command. Using single satellite doppler, time delay of arrival, and frequency delay of arrival techniques, it is possible to locate the interfering emitter. However, for the first quarter of fiscal year '93, about 50 different channels used by the Navy were non-operational for over 5000 hours. Almost 30% of the interference occurrences lasted for over 12 hours. In only one quarter of the cases was the interference turned off, either by our detection or on its own, in less than two hours.

Under the current system the old geolocation techniques and equipment is crude at best. Multiple Access Communication Satellites (MACSATs) have been used with some success in geolocation, as have Lincoln Experimental Satellite (LES) -8 and -9, but the MACSATs are becoming inoperative. They were launched in 1990 with an on-orbit design life of three years, and are experiencing difficulties. LES-8 and -9 are operated by Lincoln Laboratories, and it is not always possible to use them as needed due to schedule conflicts and since the Navy has no operational control over Lincoln Labs. A big obstacle to the current system are the difficulties in the software. Also, the antennas used both on the satellites and on the ground were not designed or optimized for the RFI mission.

It is recommended that the Navy should develop and fly a modest constellation of lightsats in order to solve the problem. It is anticipated that these satellites will cost no more than 4 million apiece to build and fly. This would save the four million a year that we currently spend on RFI, and more importantly would protect operational capability. To begin with, one small satellite could be used to demonstrate the technology. By using commercial off-the-shelf technology (COTS) and hardware, the cost of the satellite could be kept to a minimum. Launching costs would be reduced by working with the Air Force Space Test Program or other low-cost options.

With RINGSATs dedicated to the problem of RFI, the efficiency of the geolocation would be greatly increased. Satellite access concerns encountered with LES-8 and -9 would be negated. Improvements would be made in the RFI geolocation hardware by integrating Global Positioning Satellites (GPS) components to give better satellite ephemeris data and thus improve the geolocation solutions. Antennas could be optimized for the RFI problem, which would also improve the system.

In addition to solving the RFI problem, RINGSAT will also support two other missions: store and forward and GPS beacon relay. Much of the hardware used in geolocation on the satellites could also be used for the Radiant Snow (GPS based beacon) program. Position data could be relayed by the satellite to control stations on the ground. This would augment existing beacon data relay architecture enabling virtual continuous tracking of the beacons. The other secondary function of RINGSAT would be a UHF link for store and forward communications. The Navy and other services require reliable ways to pass information, and this also uses much of the equipment built into the satellite. Integrating these ideas into an RFI satellite would be a normal extension of the satellite's capabilities, and would thus be simple and low-cost.

In summary, a constellation of multipurpose lightsats would be a cost-effective solution to the RFI problem. Additional capabilities such as store and forward communications and beacon relay would fill other vital service needs. Using Naval Academy Midshipmen and Naval Research Laboratory personnel, parts and facilities to build the first satellite is a very affordable method to demonstrate and prove the operational capabilities of RINGSAT. By spending a minimal amount of money on these satellites, the Navy would significantly enhance its ability to perform RFI geolocation, increase fleet EMCON, save the Navy money, and ensure uninterrupted satellite access for the fleet.

1.1 SUMMARY

Primary Objective--demonstrate technology of a low-cost lightsat that will upon command locate the source of RFI and and relay the location to command stations

-Must pinpoint source of interference within 3 hours and be compatible with current systems
-Will be lightsat sized, octagonal and not weigh more than 300 kg, with a radius of .75 meter and height of 1 meter
-Must maintain low earth orbit for at least 5 years
-Will be in a 475 nm orbit, 14 orbits per day, with a 60 degree angle of inclination
-Attitude dynamics and pointing accuracy needed is not especially high, but must be able to transmit accurately--probably a gravity gradient boom system
-Must send data to ground upon query
-Must have high reliability
-Resolution should be within one mile
-Cost must be under eight million dollars
-Satellite must be autonomous in RFI, GPS beacon, and store and forward receiving: autonomous in store and forward and science mission transmissions, but rely on ground for RFI task and GPS beacon transmitting
-Satellite could take advantage of established Transit ground stations for sending and receiving data

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2.0 Spacecraft Requirements/Constraints/Assumptions

2.1 Requirements and Constraints

2.1.1 Orbit

* Must maintain low earth orbit for five years

* Mustbe able to locate and relay the source of interference within 3 hours (frequent revisit time)

* Satellite must effectively monitor latitudes +/- 60

* In order to meet these requirements, six satellites must be used

2.1.2 Launch Vehicle

* Must be able to insert spacecraft directly into required orbit

2.1.3 Attitude Determination and Control

* Approximate spacecraft attitude must be determined

* Must be earth - oriented

* Must be three - axis stabilized

2.1.4 Communications

* Directionalantennacoveragetosupport the Radio Frequency Interference mission.

* A capability to receive a 2 kbps command uplink from the Transit ground station to receive commands for the RFI mission.

* Acapabilitytodownlinkup to 12 kbps of stored information of RFI data to the Transit ground station.

* A capability to transpond uplink ranging signals received from the ground stations for use in the bent-pipe mission.

* On-board storage for 10 kbps of GPS beacon or bent-pipe information.

* Omni-directionalantennacoveragetosupport 1 kbps command uplink for telemetry, tracking, and control.

* Must support TOMS or other science mission

2.1.5 Structures

* Must mate with Pegasus the size requiring a radius of 1.17 meters and a height of 1.78 meters. The adapter must have one end closed to the attached mate ring

* Electronics must be thermally sound

* There must be enough surface area for solar arrays .35 m x .83 m

* Must stand up structurally to launch environment

* Must be stabl tvo proide 1 deg fpointing accuracy. Thismeans equipment must be placed properly.

* Must be enough room for all equipment

* Weight must be under 240 kilograms

2.1.6 Thermal

* Eliminate unwanted energy due to the body-mounted solar cells

* Provide a moderate temperature environment for the critical electronic, battery and fuel components

* Limit the amount of power required for the heaters to maintain temperature environment by utilizing other thermal control techniques

2.1.7 Power

* Supply a continuous source of electrical power to spacecraft loads during the mission life

* Supportpower requirements for average and peak electrical loads

* Ensureenergy storage source is capable of operating during five year mission life within the environment of space

* Ensure the energy storage source operates at the bus voltage and is fully charged during daylight

* Create decentralized power distribution system to turn

power on and off to the spacecraft loads

* Provide power regulation and control to the electrical power system

3.0 ORBITAL ANALYSIS

The RINGSAT orbit design gives the parameters for an orbit that allows the satellite to complete its mission of locating Radio Frequency Interference in the most efficient and effective manner.

3.1 REQUIREMENTS

The requirements to be met by the RINGSAT orbit are that it must:

1. maintain a low earth orbit for 5 years.

2. allow the satellite to locate and relay the source of interference within 3 hours (frequent revisit time).

3. effectively monitor latitudes +/- 60°

3.2 ORBITAL PARAMETERS (SPECIFICATIONS)

The RINGSAT orbit must be a low earth orbit that will cover considerable area of the earth in order to accommodate for mission accomplishment.

3.2.1 Semi-major Axis

The orbit of the satellite will be low earth. A low earth orbit was chosen so that RINGSAT would be able to:

* effectively use its instruments (antenna)
* cover a large area of the world with one satellite
* locate RFI at one location and then transfer the information to another location
* have several orbits a day for a quick return time

The altitude of the satellite will be 881 km (475 nm). This produces a semi-major axis of 7259 km.

An altitude of 881 km will allow for the satellite to travel 14 orbits per day. This altitude is ideal for RINGSAT because it is not too low so that it will be plagued by significant drag, and it is not too high such that the accuracy needed for the instruments is lost.

3.2.2 Eccentricity

The eccentricity of RINGSAT's orbit will be zero such that its orbit is circular. A circular orbit is the most effective orbit because:

* it allows for even coverage of the earth
* it is the simplest orbit that satisfies the satellites requirements
* gives a quick and constant return time which is paramount for locating and reporting RFI

3.2.3 Inclination

The inclination of the satellite's orbit will be 60°.

This inclination is most practical because it allows for sufficient coverage of the surface of the earth where RFI is found. RFI is not a problem in the polar regions, therefore it is not necessary for the satellite's orbit to cover this area. Figure 1 shows a ground trace of RINGSAT with its 60 inclination.

3.2.4 Period

By allowing the satellite to travel 14 orbits a day, the period of the orbit is 102.6 minutes.

$$T=\frac{2\pi}{\sqrt{\mu}}a^{3/2}=\frac{2\pi}{\sqrt{3.986x10^5}}7259^{3/2}=102.6mins$$

During each orbit, the earth rotates 25.71° and therefore after 14 orbits, the satellite returns to the same location it started in. This period is very appropriate because it allows the satellite to recover similar areas of the earth in approximately 3 hours.

By complementing one RINGSAT satellite with five other satellites in the same orbit, the affected area of the Earth will be entirely covered and the mission of RINGSAT will be accomplished in the most timely manner.

3.3 ORBITAL ANALYSIS SUMMARY

The RINGSAT orbit has the following specifications:

* Semi-major Axis (a) = 7259 km
* Eccentricity (e) = 0
* Inclination (i) = 60°
* Period (T) = 102.6 min
* Orbital Decay = -.0473 km/yr

4.0 LAUNCH VEHICLE

4.1 REQUIREMENT

The requirement for the launch vehicle is that it must:

1. be able to insert spacecraft directly into required orbit.

4.2 PEGASUS LAUNCH VEHICLE

The Pegasus vehicle is the best suited launch vehicle for RINGSAT. Although the only functional requirement for the launch vehicle is direct orbital insertion, the other aspects of the Pegasus make it the ideal vehicle to launch one RINGSAT satellite.

4.2.1 Primary Mission

Pegasus is capable of launching a light-weight satellite into any low earth orbit at any inclination.

The Pegasus is carried by a conventional transport/bomber-class aircraft (B-52 or L-1011). Due to the fact that it is air-launched, its launch azimuth is from zero to 360°.

4.2.2 Launch Site

The Pegasus' primary launch site is Vandenberg Air Force Base in California. The location of this launch site is optimum for a 60° inclination.

4.2.3 Performance

For an 881 km, circular orbit, the Pegasus launch vehicle can carry a payload up to 246 kg. Due to the size and weight of RINGSAT, only one satellite may be launched at a time. The Pegasus XL will use three stages to insert the satellite directly into its orbit. Upon orbital insertion, the satellite is expected to have little or no spin rate.

4.2.4 Payload Fairing Size

The fairing envelope of the Pegasus has a diameter of 1.27 meters and a height of 4.43 meters. RINGSAT, with its diameter of 0.7 meters and height of 1.0 meter, will comfortably fit.

4.2.5 Cost

The estimated launch price for each Pegasus is between 7 and 12 million dollars. Therefore, this is the most costly portion of the RINGSAT budget.

5.0 ATTITUDE DETERMINATION AND CONTROL

The attitude determination and control demands of RINGSAT are not extensive, but it is imperative that the attitude of the spacecraft be determined during orbital insertion/acquisition. During this time, the satellite will be oriented into the proper attitude to deploy a gravity gradient boom.

5.1 REQUIREMENTS

The requirements for RINGSAT's attitude control system are:

1. to determine approximate spacecraft attitude

2. to orient the spinning/tumbling satellite into an attitude appropriate for deployment of gravity gradient boom

3. to maintain satellite in an Earth orientation

4. to ensure the satellite remains three-axis stabilized

5.2 GRAVITY GRADIENT BOOM

A satellite with a gravity gradient boom is the best and cheapest choice for keeping an antenna pointed at the Earth at all times.

The antenna planned for use on RINGSAT has a very high gain, therefore the pointing of the spacecraft does not need to be very accurate (+/- 10°). A typical gravity gradient boom has a pointing accuracy of +/- 20° but it is significantly improved by the addition of an eddy damper as the tip mass. This brings the accuracy into the range of +/- 1°.

5.3 ATTITUDE DETERMINATION

At orbital insertion, RINGSAT will likely be spinning or tumbling, therefore in order to determine its attitude, RINGSAT will have 6 sun sensors and a magnetometer. These provide the simplest and least expensive means of determining attitude.

5.3.1 Sun Sensor System

In order to ensure that a sun sensor is always facing the sun while not in eclipse, it is necessary to place several sensors around the body of the spacecraft. Obviously the sun sensors are not of any use while RINGSAT is in eclipse, but during this time the attitude can be adequately estimated by the magnetometer alone. The sun sensor system can accurately determine attitude to within 0.005° - 3.0°.

5.3.2 Magnetometer

A magnetometer is inexpensive and significantly accurate enough to use as another means of attitude determination. Based on a knowledge of the Earth's magnetic field, it can determine the spacecraft attitude to within 0.5°.

5.4 ATTITUDE CONTROL

The gravity gradient boom is the main attitude control system for the spacecraft.

During acquisition, the spacecraft needs to be correctly oriented in order to deploy the boom. Eight small cold gas (N_2) thrusters will be used for this purpose. These will be symmetrically located on the edge of the spacecraft and positioned to fire around the center of mass.

6.0 COMMUNICATIONS SUBSYSTEM

The RINGSAT communications subsystem gives the capacity to receive the command uplink from the ground, to listen on the interference frequency indicated to geolocate the source, to "bent pipe" information from the onboard antenna to the ground stations, and to provide housekeeping telemetry data that will allow the ground station to see how the satellite is functioning.

6.1 REQUIREMENTS

The requirements to be met by the RINGSAT communications subsystem are to provide:

1. Directional antenna coverage to support the Radio Frequency Interference mission.

2. A capability to receive a 2 kbps command uplink from the Transit ground station to receive commands for the RFI mission.

3. A capability to downlink up to 12 kbps of stored information of RFI data to the Transit ground station.

4. A capability to transpond uplink ranging signals received from the ground stations for use in the bent-pipe mission.

5. On-board storage for 10 kbps of GPS beacon or bent-pipe information.

6. Omni-directional antenna coverage to support 1 kbps command uplink for telemetry, tracking, and control.

6.2 DESIGN

The principal design driver is the mission requirement. This is to find and downlink the desired Radio Frequency Interference location data information in a ten minute pass to the ground station. The existing Transit ground stations have equipment to receive the transmission with no problems. Table 1 and Figure 1 show expected link margin calculations for the forward and return links for a bit error rate of 10^{-5}. Link margins are given for the maximum range or worst case of the antenna beamwidth.
The gravity gradient design of RINGSAT allows the effective use of the main antenna which is a crossed turnstile dipole antenna. Since the main mission is to have the antenna pointing at the earth at all times, the gravity gradient stability system is perfect for the mission. The crossed turnstile dipole antenna was chosen over a helix or omnidirectional antenna for the reason that maximum gain could be acheived looking towards the earth with this beam pattern. The object was to obtain the most area coverage with the gain that was given. This antenna provides maximum coverage for the slant ranges whereas the helix antenna would have a circular beam pattern and therefore not as much area coverage with the same gain. The omnidirectional antenna would waste most of its gain looking into space.

The design of the RINGSAT communications subsystem has the principal components of: 6 S-band transponders, 6 band reject filters, 6 diplexers, one DSP-12 digital signal processing controller, four omni-directional antennas, one turnstile crossed dipole antenna, and 4 megabytes for information storage.

For a data rate of 16 kbps, a 263 MHz signal, with a signal to noise ratio of 30 dB the power transmitted is 12.97 watts. This gives an acceptable bit error rate of 10^{-5} at the maximum distance in the antennas beamwidth. Binary phase shift keying (BPSK) will be the method of modulation used. In BPSK, the digital data modulates a sinusoidal carrier, as illustrated in Figure 2. Most of the energy in the modulated signal is found in the main lobe. The minor lobes repeat indefinitely through the spectrum. However, an additional technique called minimum-shift keying (MSK) can be used to artificially put more of the power into the main lobe, and therefore the energy in the sidelobes is not needed. It is the simplest form of phase shift keying, has good use of the spectrum and a relatively low bit error rate, and is much cheaper and uses less energy than quaternary phase shift keying (QPSK). QPSK has better use of the spectrum, but that is not needed if MSK is used.

Forward error correction coding is also not needed, due to a relatively low bit error rate. Also, if redundancy bits were added in either a block or convolutional code system, the bit rate itself would need to be increased. Using the current system, this would be possible to do since the margin is as high as 13 dB. However, it is anticipated that the current error is acceptable and that the data downlinked would be better spent on a larger number of users for the system.

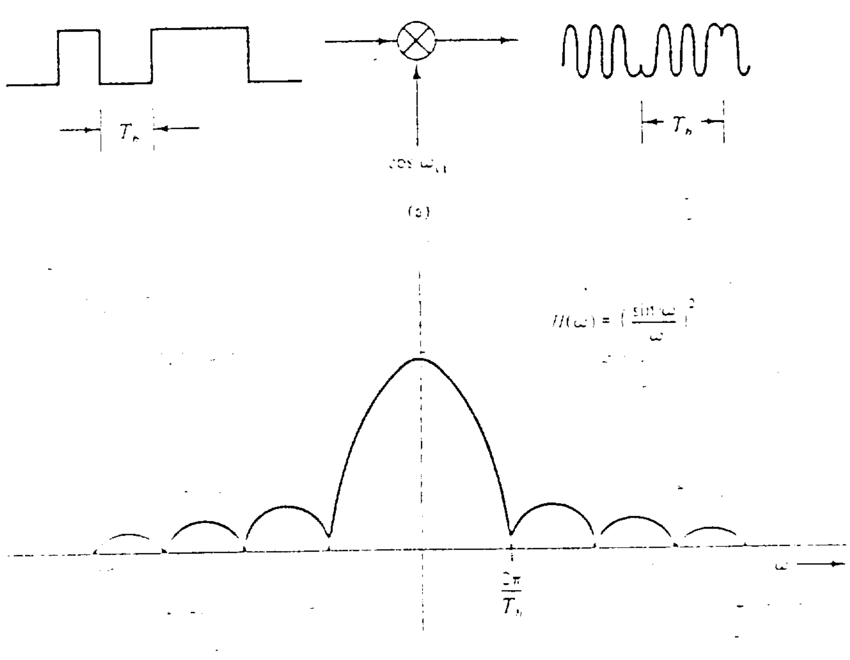


Figure 2 — Binary phase-shift keying (BPSK): (a) time domain; (b) frequency domain.

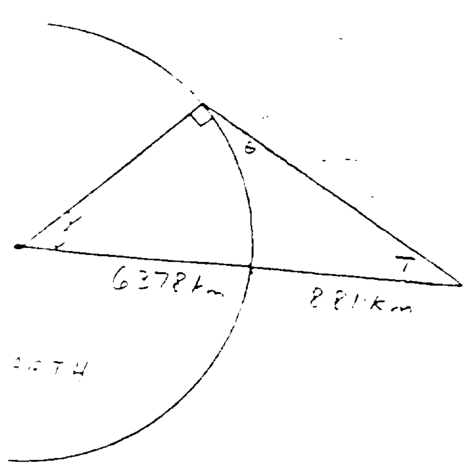
TABLE 6.1

Item	Symbol	Downlink	Uplink	Remarks
Frequency	f (MHz)	263	273	Input
Line loss	L_l (dB)	-3	-3	Standard
Transmit Antenna Beamwidth	θ_t(deg)	91.287	7.5	Input
Peak Transmit Antenna Gain	G_{pt} (dB)	5.0918	26.7988	Eqn. 1
Lgth/Dia Transmit Antenna	L/D_t (m)	0.75	18.30	Eqn. 2
Transmit Antenna Point Offset	e_t (deg)	10.00	20.00	Standard
Transmit Antenna Pointing Loss	L_{pt}(dB)	-0.14	-85.33	Eqn. 3
Transmit Antenna Gain	G_t (dB)	4.95	-58.53	Eqn. 4
Maximum Slant Range	S (km)	1368.12	1368.12	Fig. 1
Space Loss	L_s (dB)	-83.56	-83.89	Eqn. 5
Lgth or Dia Receiver Antenna	L/D_r (m)	18.30	0.75	Given
Peak Receive Antenna Gain	G_{pr} (dB)	25.00	5.09	Given
Receive Antenna Beamwidth	θ_r (deg)	7.50	91.29	Given
Receive Antenna Pointing Error	e_r (deg)	20.00	10.00	Given
Receive Antenna Pointing Loss	L_{pr} (dB)	-85.33	-0.14	Eqn. 3
Receive Antenna Gain	G_r (dB)	-60.33	4.95	Eqn. 4
System Noise Temperature	T_s (K)	375.00	1295.00	Standard
Data Rate	R (bps)	16000.00	4000.00	Input
Bit Error Rate	BER (N/A)	0.00001	0.00001	Input
Required Signal to Noise Ratio	E_b/N_o(dB)	14.40	14.40	Input
Desired Signal to Noise Ratio	E_s/N_o(dB)	30.00	21.00	Input
Power Transmitted	P_t (dBW)	11.13	0.02	Eqn. 6
Power Transmitted	P_t (W)	12.972	1.005	Input
Equiv Isotropic Radiated Power	EIRP(dBW	13.08	-61.52	Eqn. 7
Margin	(dB)	13.60	4.60	Eqn. 8

Input means that the requirements of the satellite dictated the number.
Standard means that it is a standard value for that variable.
Given means that the equipment to be used is already set at that number.

FIGURE 6.1

Theta = elevation angle, T = nadir angle, gamma = earth angle



$$T = 1/2\ Beamwidth = 45.6435^\circ$$

$$\sin T = \frac{R_e}{r}\cos\theta$$

$$\theta = 35.5341^\circ$$

$$\gamma + \theta + T + 90^\circ = 180^\circ$$

$$\gamma = 8.8224^\circ$$

$$Slant\ range = \frac{R_e \sin\gamma}{\sin T}$$

$$Slant\ range = 1368.1157\ km$$

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Equations for the Link Budget

$$\text{Eqn 1:} \quad G_{p_t} = 44.3 - 10 * \log(\theta^2)$$

$$\text{Eqn 2:} \quad Length = (5/8) * (c \,/\, frequency)$$

$$\text{Eqn 3:} \quad L_{p_t} = -12 * (\frac{e}{\theta})^2$$

$$\text{Eqn 4:} \quad G_t = G_{p_t} + L_{p_t}$$

$$\text{Eqn 5:} \quad L_s = 20 \log c - 20 \log(4\pi) - 20 \log S - 20 \log f$$

$$\text{Eqn 6:} \quad P_t = \frac{E_b}{N_o} - L_l - G_t - G_r - 228.6 + 10 \log T_s + 10 \log R$$

$$\text{Eqn 7:} \quad EIRP = P_{dBW} + L_l + G_t$$

$$\text{Eqn 8:} \quad Margin = Desired\frac{E_b}{N_o} - Required\frac{E_b}{N_o} - 2$$

6.3 SUBSYSTEM DESCRIPTION

The communications subsystem interfaces with the C&DH subsystem and the power subsystem. The turnstile crossed dipole-antenna receives demodulated signals from the interferer on the ground and this information is passed to the C&DH, and the C&DH sends this to the transmitter for downlinking to the Transit ground station. Figure 3 shows a block diagram of the communications subsystem.

6.3.1 Transponders

The transponder includes the following components:

* A transmitter with a modulator and a telemetry encoder
* A receiver with a demodulator and command decoder
* Monitor and control system

There will be six transponders on RINGSAT with the downlink frequencies of 250-311 MHz and the uplink frequencies of 250-311 MHz, which is the military UHF range, and they will both have a military standard bandwidth of 25 kHz. There will be one uplink for the omnidirectional antennas and two downlink frequencies for information and telemetry and control. An onboard GPS receiver will be used for precision tracking during the RFI mission. In addition, there will be two beacons aboard as a fail-safe system if everything else goes wrong with the satellite.

6.3.2 Omni-directional Antennas

Four omni-directional antennas wil be mounted to RINGSAT, all on the center of mass and all perpendicular to each other. This is to provide full coverage in all directions. There will have to be a deployment mechanism to extend the antennas from the body.

6.3.3 Turnstile Crossed Dipole Reflector Array Antenna

The main mission of the RINGSAT spacecraft will primarily use the turnstile crossed dipole antenna for receiving the radio interference. This antenna has the advantage of being very inexpensively built and it can be put into operation without the need for test equipment. The length of the antenna is 75 cm. This antenna consists of four radiating elements mounted on the -Z surface (bottom) of the spacecraft. The signal produces a toroidal pattern. The elements of the antenna are made of flexible, springy, semi-cylindrical metal approximately 1.0 cm in width. The distance between the satellite plate mount and the antenna dipole is 5/8*wavelength. The Beamwidth is 91.287 degrees. This antenna gives us a gain of about 5.0.

6.3.4 RFI Geolocation

Upon receiving the command to geolocate on a certain frequency from the Transit ground station, the satellite would listen in the area of the frequency indicated for the interfering signal. Upon finding the signal, it would begin collecting data on the frequency, time, and position of the satellite until it loses the signal. Then, the data handling system on board the satellite would graph the frequency versus the time for the signal, as shown in Figure 4. From the graph, the second derivative is taken of the curve to get the point of inflection, which is the closest point of approach (CPA) to the interferer. Then, the slope at that point is found, and that is the value of m in the derivation that follows. The final derived equation solves for d, which is the slant range to the source of the interference. Since the satellite is in a low earth orbit, it will only have one closest point of approach to the interferer during any given pass, which means the interferer is on a line 90 degrees perpendicular to the ground trace. Since the position of the satellite at that point is known, and the distance to the interferer is known, the source of interference is located to be at one of two points, as shown in Figure 5. The slant range d and the position of the satellite at the CPA time is then sent to the next ground station.

Derivation of the slant range equation

$$\vec{r}_s = vt\hat{i} + \rho_o \hat{j}$$

$$\vec{r}_o = 0\hat{i}$$

$$\vec{v}_s = v\hat{i}$$

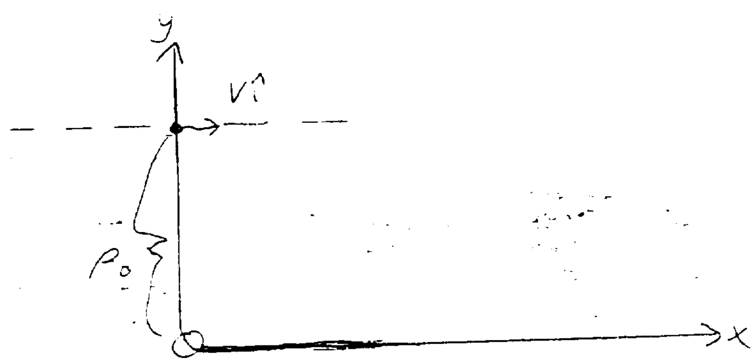
$$f = f_o \frac{\sqrt{1 - v^2/c^2}}{1 - \dfrac{\vec{v}_s \cdot \vec{r}_{so}}{c\ r_{so}}}$$

Gives doppler shift, taking relativity into account

$$\vec{r}_{so} = -vt\hat{i} - \rho_o \hat{j} = \vec{r}_o - \vec{r}_s$$

$$\vec{r}_{so} = \sqrt{v^2t^2 + \rho_o^2}$$

$$\vec{v}_s \cdot \vec{r}_{so} = -v^2 t$$



$$f = f_o \frac{\sqrt{1 - \dfrac{v^2}{c^2}}}{1 - \dfrac{v^2 t}{c\sqrt{v^2t^2 + \rho_o^2}}}$$

$$\frac{df}{dt} = -f_o\sqrt{1-\frac{\nu^2}{c^2}}\,\frac{\dfrac{-\nu^2}{c\sqrt{\nu^2t^2+\rho_o}} + \dfrac{\nu^2t(2\nu^2t)}{2(\nu^2t^2+\rho_o^2)^{3/2}}}{\left(1+\dfrac{\nu^2t}{c\sqrt{\nu^2t^2+\rho_o^2}}\right)^2}$$

$$\frac{df}{dt} = \frac{-f_o\nu^2}{c\rho_o}\sqrt{1-\frac{\nu^2}{c^2}} = m$$

$$\rho_o = -f_o\nu^2\frac{\sqrt{1-\frac{\nu^2}{c^2}}}{cm} = \frac{-f_o\nu^2}{cm}$$

Where the rho$_o$ equals d, which is the value of the slant range, or the distance between the satellite and the interferer. The $-f_o$ is equal to the frequency that the interferer is transmitting on, ν is the velocity of the satellite, c is the speed of light, and m is the slope of the line at the point of inflection.

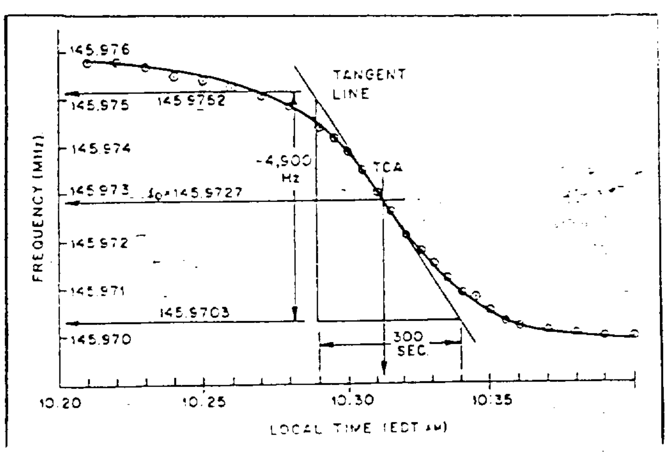


Fig. 4 —AMSAT-OSCAR 7 Doppler curve, orbit 7603, 14 July 1976, as observed from Baltimore, MD. Using the triangle shown, we evaluate the slope at TCA: m* = (−4900 Hz)/(300 sec) = −16.3 Hz/s. The satellite velocity was determined in Sample Problem 11.4, v = 7.13 km/s. Applying Eq 13.4, we obtain the slant range at closest approach, ρ_o = 1520 km.

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6.4 SUBSYSTEM RESOURCE SUMMARY

Table 2 shows the size, weight, power, and thermal requirements for RINGSAT communications subsystem. The transponders are the only active devices in this subsystem, with each using 5 watts for a total maximum of 28 watts. The maximum power transmitted for downlink is 12.97 watts. Using the average daily contact time of 140 minutes the transmitter would be turned on for 140 minutes and therefore use 1.261 watts of power per hour. The DSP-12 will also need 11.25 watts of power to handle and control the data.

6.5 SUBSYSTEM OPERATION

When is the transmitter turned on it will be left on for the full time that it is within transmission communications with the ground station. This time has been calculated to be approximately 140 minutes. The beacons are used to provide a reliable, strong downlink for the ground station so that they are able to follow RINGSAT's daily progress. There are two beacons onboard, the engineering beacon, which is mainly operated when the omni-directional antenna is used or in case of emergency, and the general beacon, which is operated continuously. This provides redundacy for reliability. These beacons can be used for telemetry and control, store and forward, and Doppler shift studies.

Telemetry functions are managed by an onboard computer and is downlinked over regular beacon or transponder downlink channels in packets. The telemetry system allows for 200 telemetry points.

6.6 GROUND STATION OPERATION

The tracking facilities for the RINGSAT spacecraft are as follows:

Laguna Peak, California
Prospect Harbor, Maine
Rosemount, Minnesota
Wahiawa, Hawaii
Howard County, Maryland
Point Mugu, California

Using this system, it will be easy to demonstrate the technology of the RFI and GPS beacon or bent pipe mission. All the antennas in place will be able to transmit and receive the information. Special training and additional software will need to be put into place, and a system for sending the information to the ground users needs to be provided. When the full constellation is built, more ground stations could be added as needed to receive information in the gaps, such as the southeast coast.

7 Structures

7.1 Weight

The overall weight of the structure is 199.176 Kg. This is under the 240 Kg weight limit for the Pegasus launch system. the weight budget is shown in Table 7.1.

7.2 Shape

The satellite is a prolate octagonal structure having height 1.17 m and diameter, side to side, of .845 m and corner to corner of .9146 as shown in Fig. 7.1. These dimensions must conform with the area the Pegasus launch vehicle provides as shown in Fig. 7.2. The structure will be made of aluminum, which will provide the main support. An octagonal shape was used for several reasons. It maximizes the space inside the electronics compartment. The square sides of the spacecraft can be easily and more cheaply made. They are easily worked with, providing more simple access to the inside. The Sandwiched Honeycomb material used on the outside on which the solar cells are mounted are not made in other than flat plates or boxes. The Sandwiched Honeycomb material provides a high strength to weight material as well as good thermal properties.

The structure consists of three chambers. The two end chambers are open at their respective ends, with the exception of the one providing the interface to the Pegasus launch vehicle. This end has a sandwiched honeycomb plate with the middle hollowed out (internal radius of .5 m) in order to provide an exit for the deployment of the gravity gradient boom. The Pegasus ring Shown in Fig 7.3 and Fig 7.4 will be attached with 32 equally spaced bolts. This connection will be done by the company providing the launch vehicle.

The other end has a cross bar support on the end for the dipole antenna in order to better provide for its structural fitness. These two end cambers will be enclosed with Honeycombed material on each of the eight sides on which the solar panels will be mounted. Each end chamber has a height of .415 m. These chambers will house no electronics due to thermal considerations.

The Middle chamber will house all of the electronic equipment to meet thermal requirements. The eight sides of this chamber will be covered by vents to provide enough cooling for the equipment. This chamber has a height of .3 m. The shelves making up either end of this chamber will be honeycombed material. Fig 7.5, an artist conception of the spacecraft shows this middle chamber. The equipment will be secured using the various methods shown in Fig. 7.6. The equipment will fit onto the bottom shelf as shown in Fig 7.1.

Table 7.2 shows the axial center of gravity location on the X axis as allowed by the Pegasus. Table 7.3 shows the error tolerance for all three axes as well as that of the mass and moment of inertia.

7.3 Deployed equipment

The deployed equipment will consist of the Gravity Gradient Boom and four antennas for Omni Direction. The 18.288 m Gravity gradient boom will be deployed via a small electric motor. An eddie damper will serve as the tip mass. The four antennas will be spring loaded and will deploy after the gravity gradient boom.

7.4 Structural Integrity

The structure must stand up to the rigors of pre-launch, launch, and final on station. The most strenuous of these is the launch environment. During launch the Satellite will be subjected to linear acceleration in the x,y and z direction. These acceleration are shown in Table 7.4, Table 7.5 and Table 7.6. The satellite can be modeled as a cantilever beam with a distributed load acting upon it for the y and z direction. For the x direction acceleration this is a compressive force and each chamber must be examined separately. Also the Sandwiched Honeycomb interface plate connected to the Pegasus mating ring will be modeled as a simply supported plate. The acoustic environment is shown in Table 7.5, Table 7.6 and Table 7.7.

Item	m(kg)
Struc	50
Trans1	10
Trans2	5
Trans3	5
Trans4	5
Trans5	5
Trans6	5
G.G boo	4
G.G. wt	5
Dip Ant	0.5
Dip ant	0.5
Dip Ant	0.5
Bl Box	20
Mag box	1
Pow Bus	2
Pow bus	2
Pow bus	2
thrusters	4
fuel	2.5
Omni 1	4
Omni 2	4
Omni 3	4
Omni 4	4
Battery	13.026
Peg adap	2
Peg ring	3
shelf 1	2
shelf 2	2
shelf 3	2
sun sen 1	0.2
sun sen 2	0.2
sun sen 3	0.2
sun sen 4	0.2
sun sen 5	0.2
sun sen 6	0.2
s.s. Box1	0.5
GPS rec	4
GPS Rec	4
TOMS	20
total	198.726

Table 7.1 (All weights in Kg)

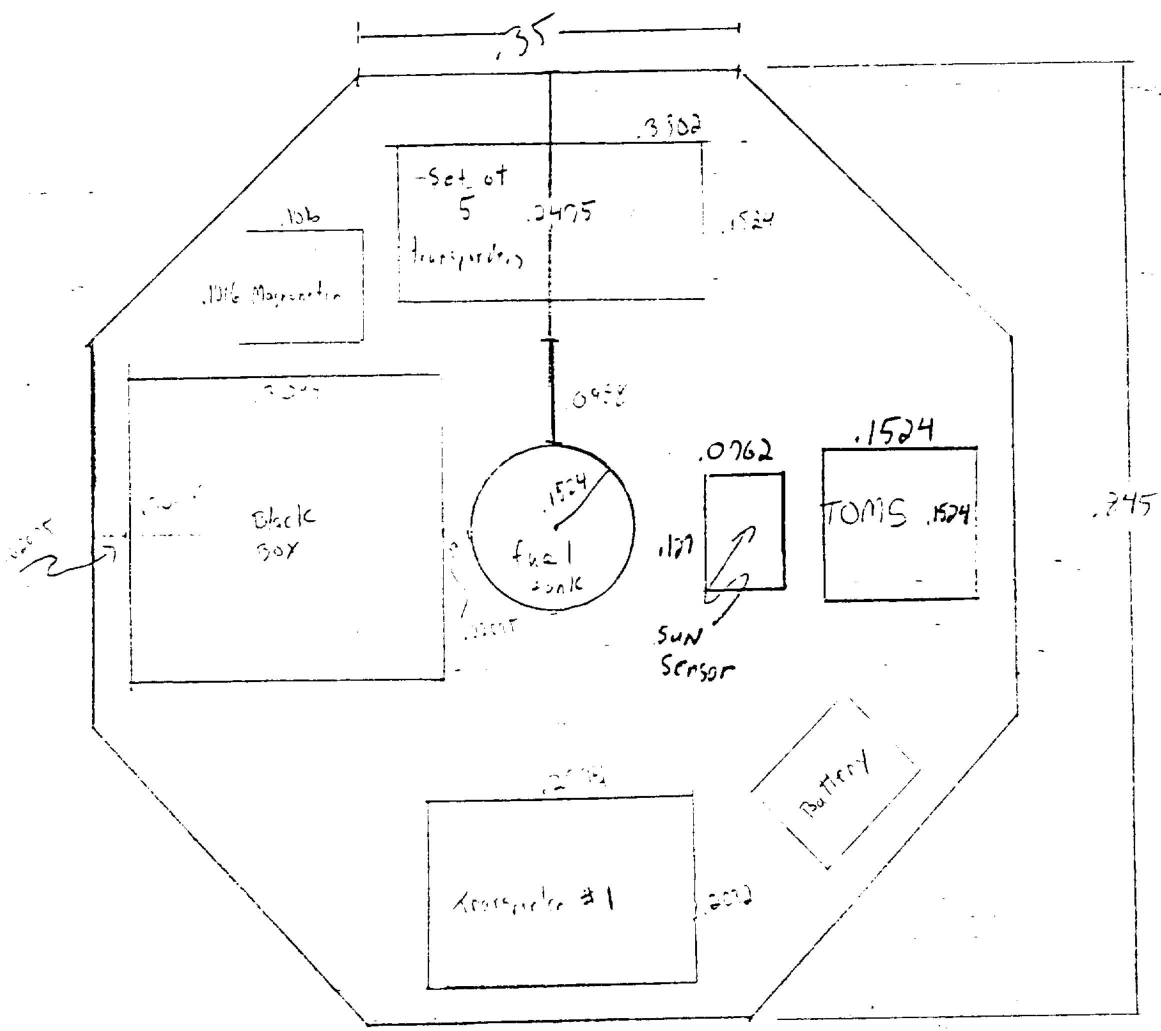

FIG 7.1 (ALL MEASURMENTS IN METERS)

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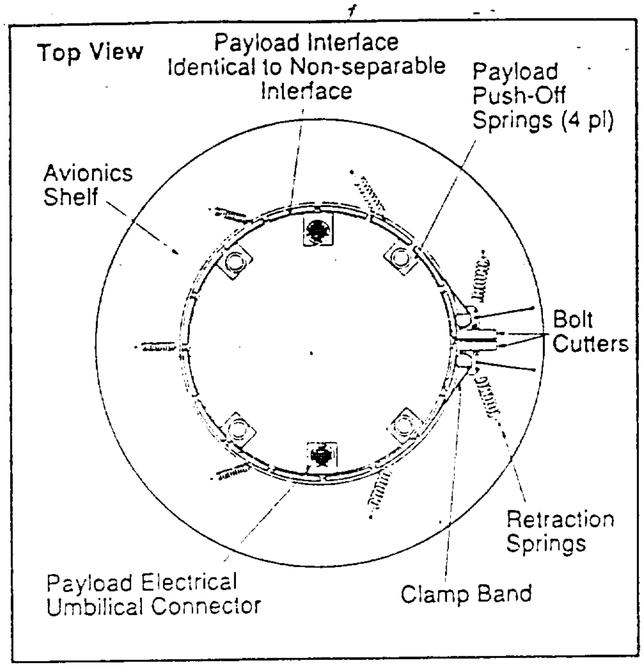

Separable payload interface.

FIG 7.3

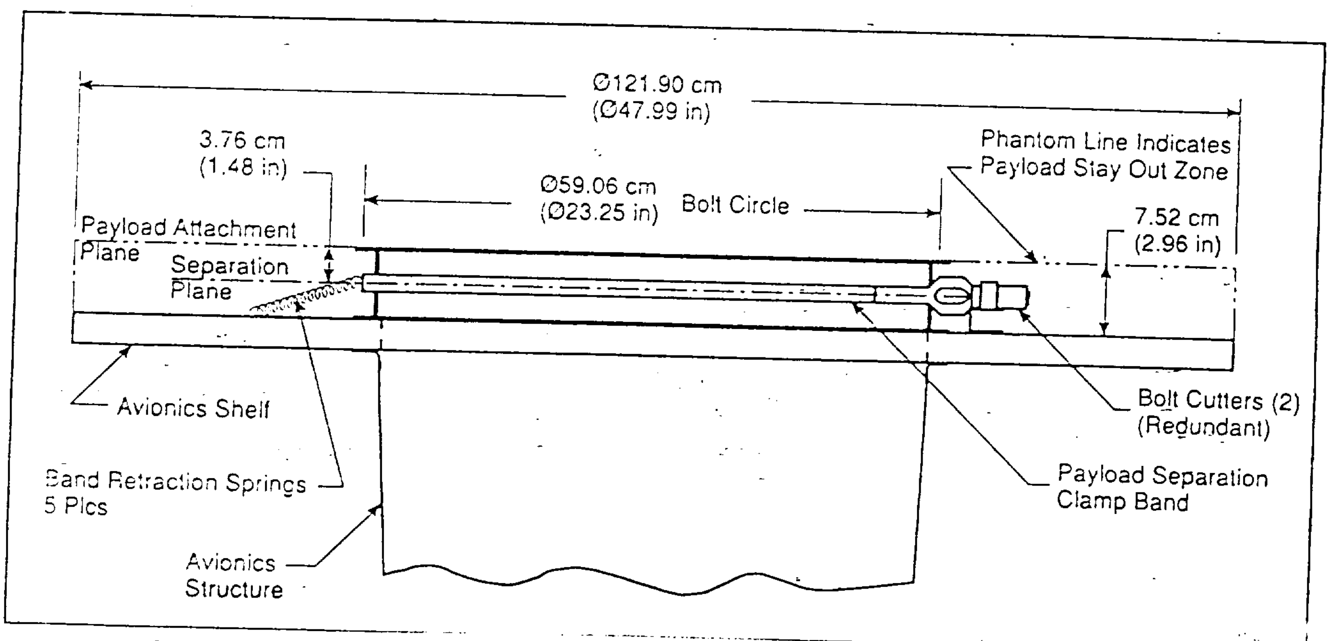

FIG 7.4 *Separable payload interface - side view.*

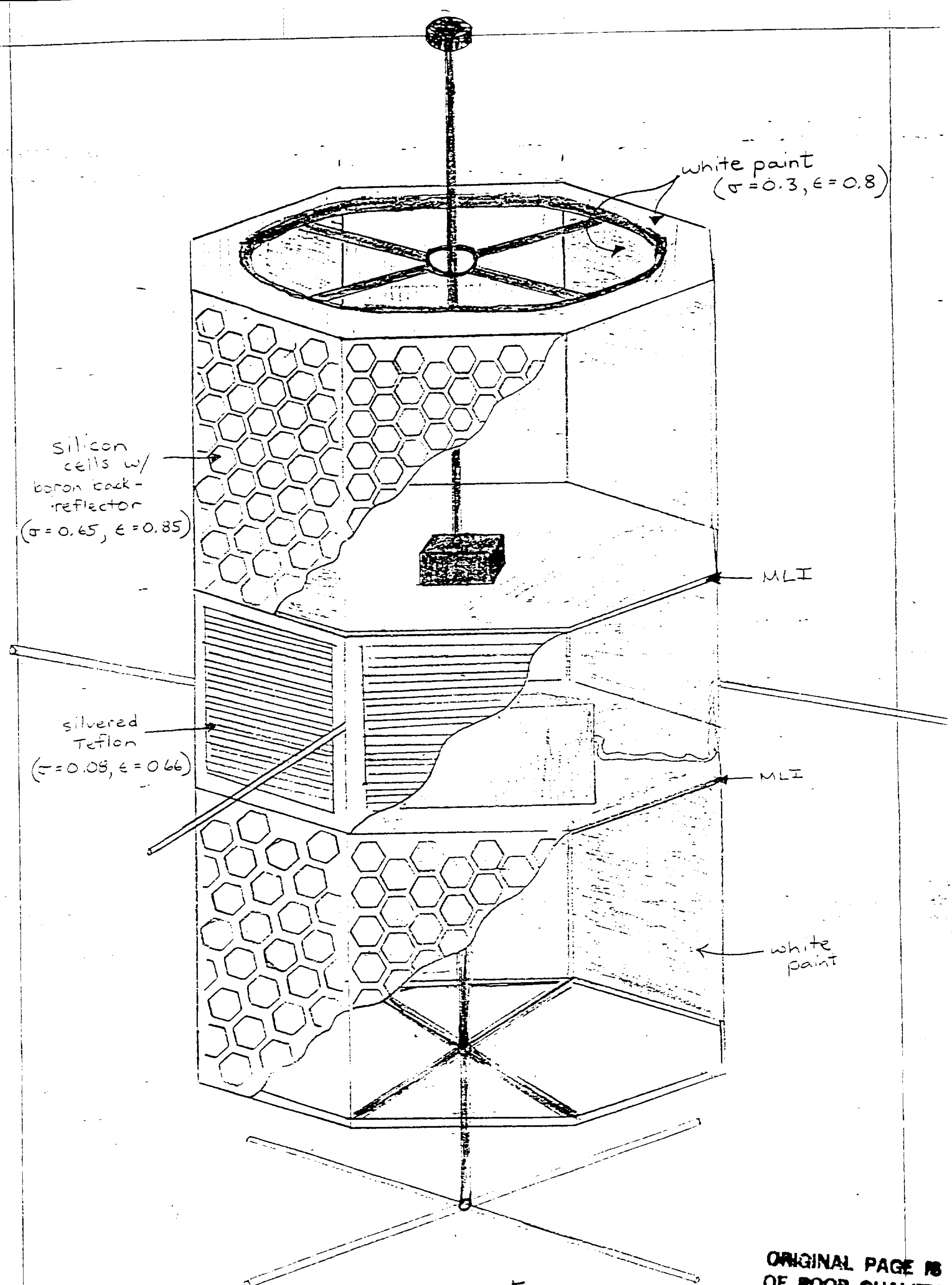
white paint
(σ = 0.3, ϵ = 0.8)
silicon
cells w/
boron back-
reflector
(σ = 0.65, ϵ = 0.85)
MLI
silvered
Teflon
(σ = 0.08, ϵ = 0.66)
MLI
white
paint

Fig 5

Figure 14

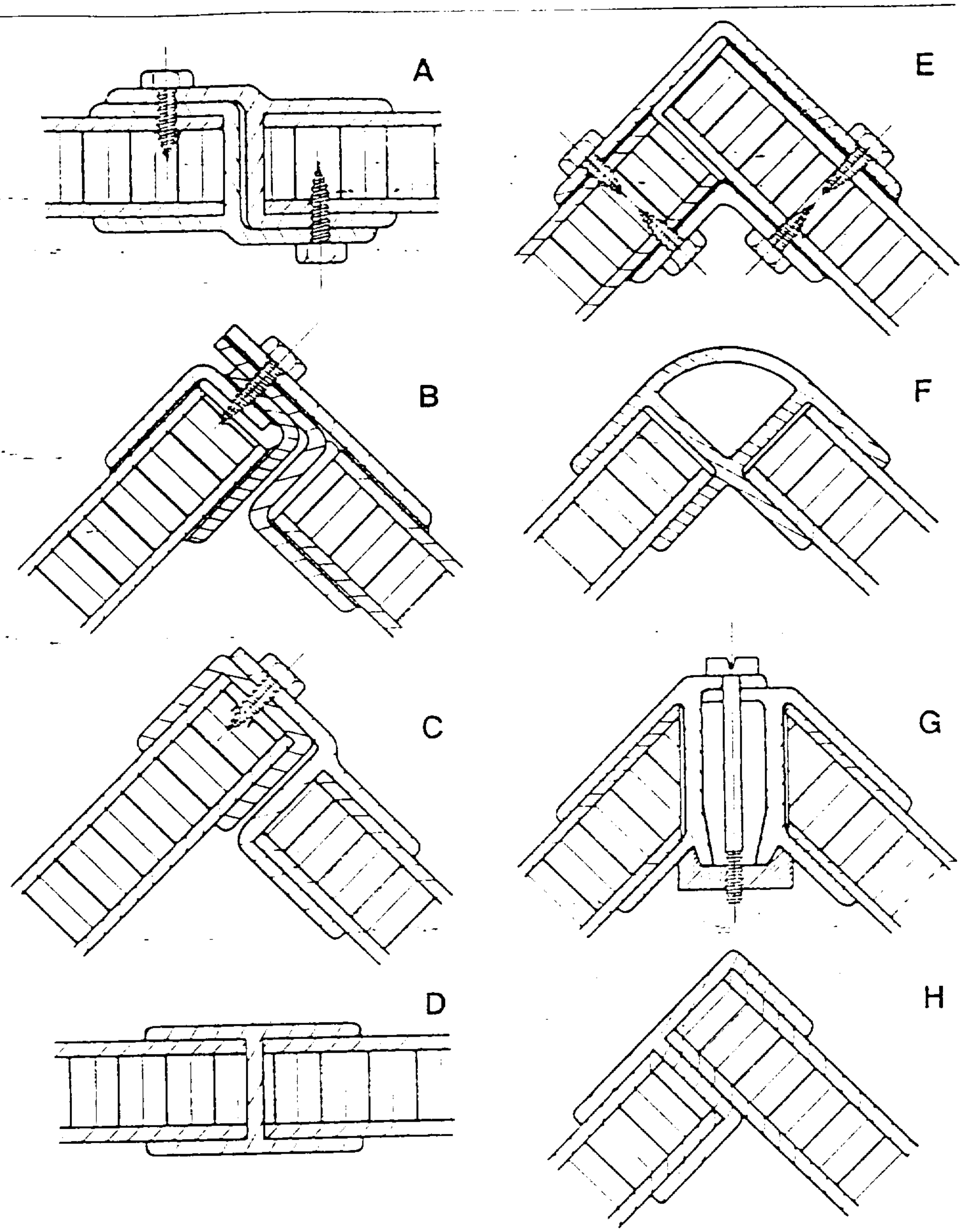

Figure 15

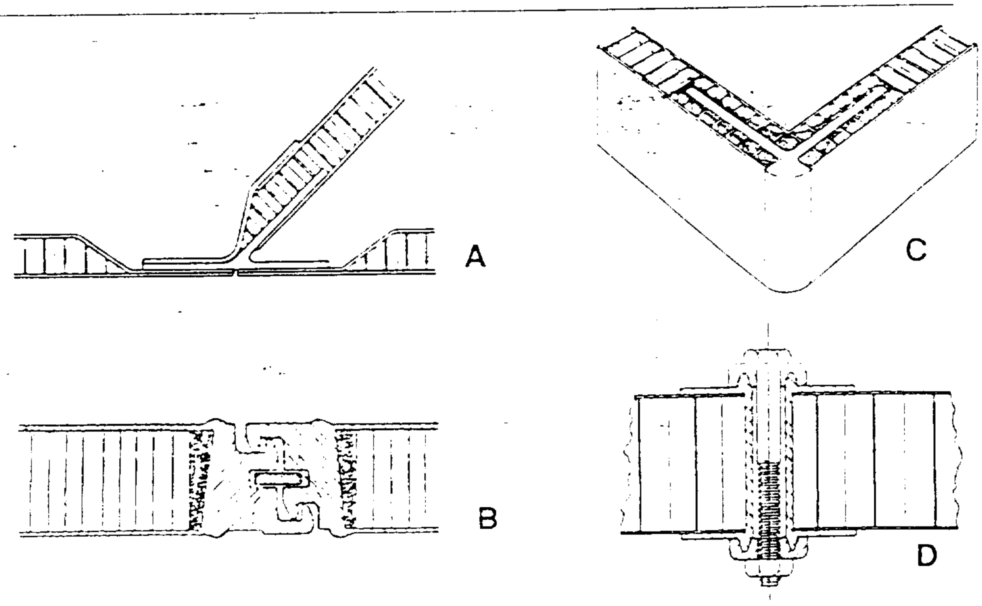

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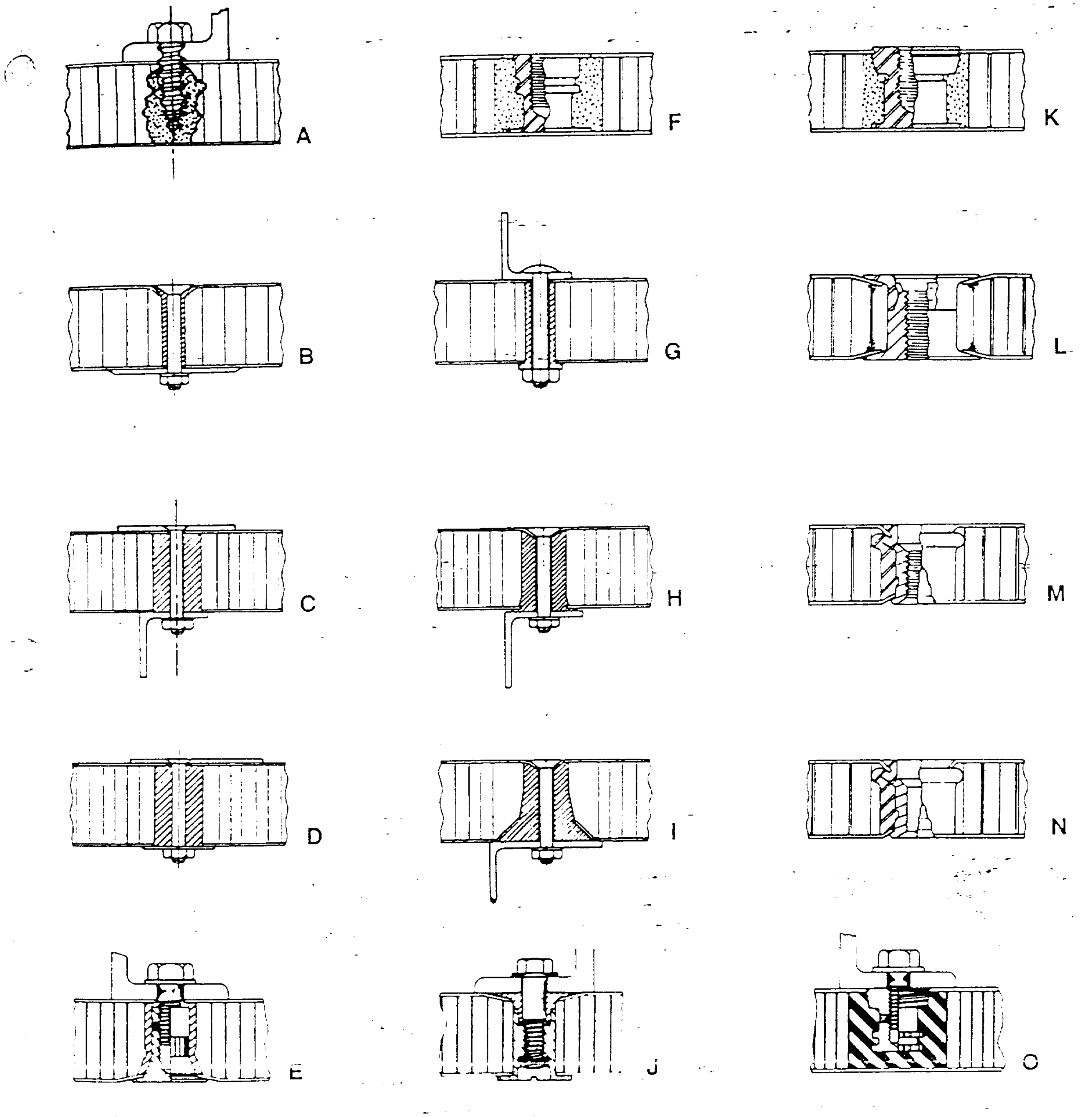
Figure 20



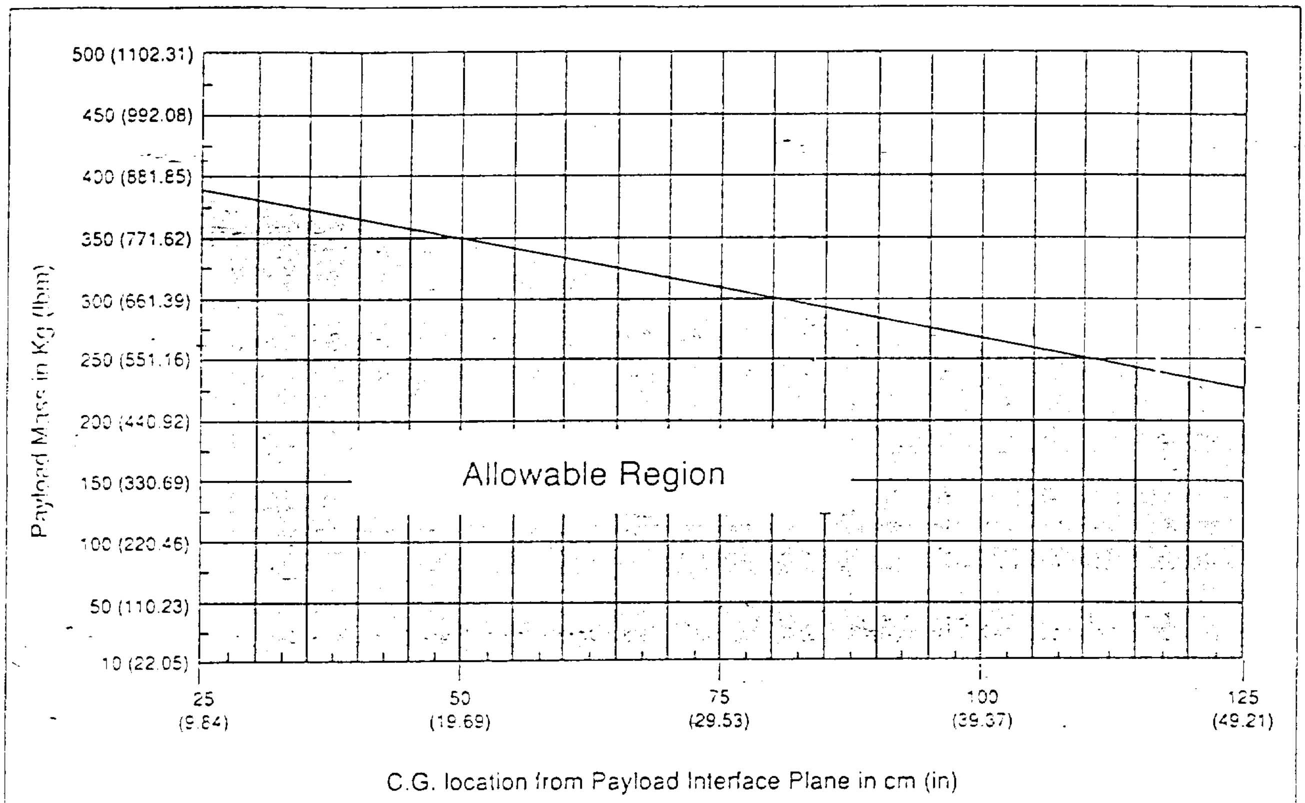


TABLE 7 2 *Payload mass vs. axial cg location on X axis (applicable to both the standard and HAPS configurations).*

TABLE 7 3 *Payload mass property measurement error tolerances.*

Measurement	Error Tolerance	
	With Standard Pegasus	With HAPS
Mass	±.5%	±0.5%
Moments of Inertia	±1%	±1%
Center of Gravity Y, Z Axes X Axis	 ±13 mm (±0.5 in) ±13 mm (±0.5 in)	 ±6.4 mm (±0.25 in) ±13 mm (±0.5 in)

TABLE 7.4 *Payload acceleration environment*

Type		Linear Acceleration (g's)		
		x	y	z
Ground Operations		+/-0.5	+/-0.5	+ 1.5
Captive Carry Flight/Taxi [1]	Case 1	+/-1.0	0	+ 2.2/-1.0
	Case 2	0	+/-0.3	+ 2.2
	Case 3	0	+/-0.4	+ 1.0
Abort Landing		+/-0.6	+/-0.6	+ 2.7/-0.1
Launch/Drop		0	0	Payload Dependent
Aerodynamic Pull-up		-4.0	+/-0.5	+2.85
First Stage Burnout		-7.5	+/-0.2	+/-0.2
Second and Third Stage Burnout		Payload Dependent (See Figure 4.1)		

Note 1: Payload must meet each of these case requirements individually to satisfy NASA-Dryden B-52 utilization constraints

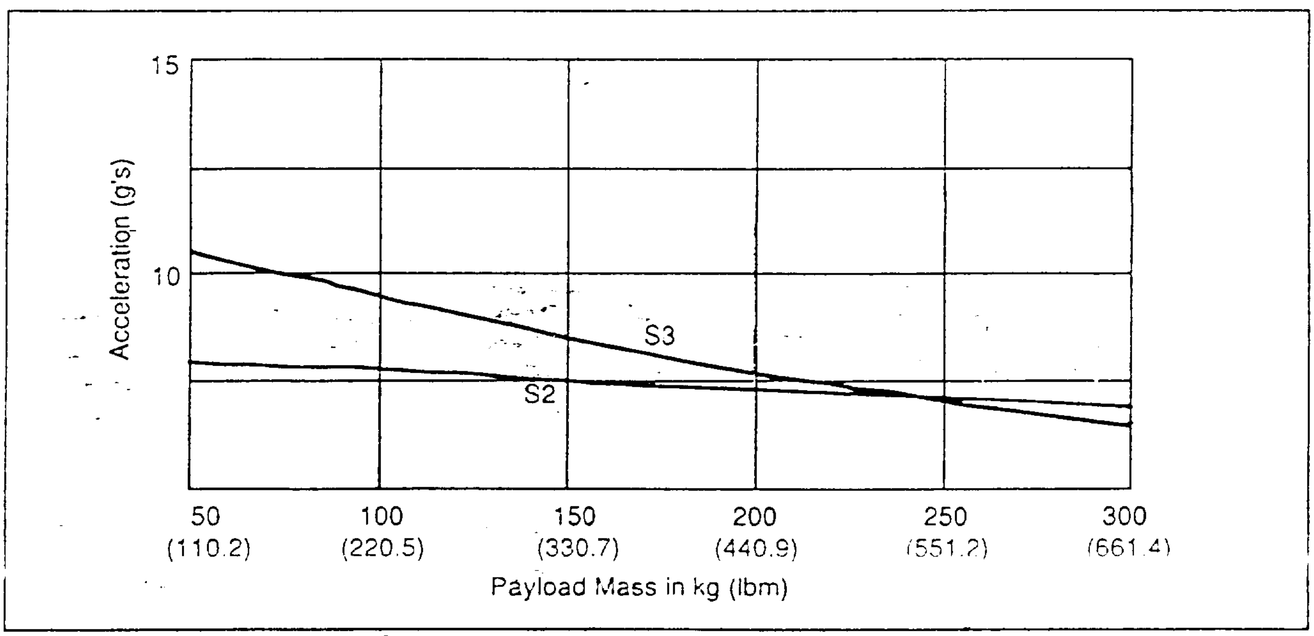


TABLE 7.5 *Payload maximum axial acceleration.*

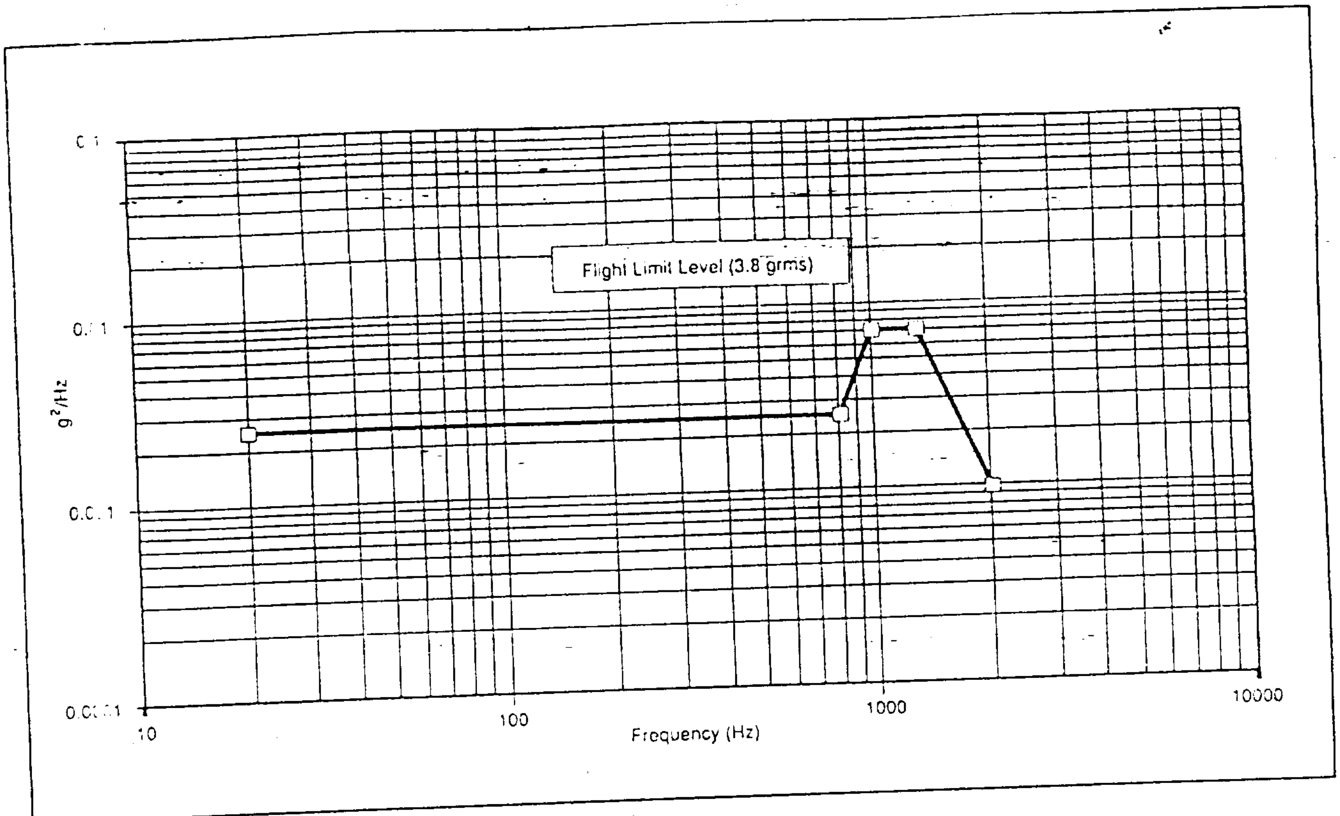


TABLE 7.5. *Payload random vibration environment.*

Pegasus® Payload User's Guide

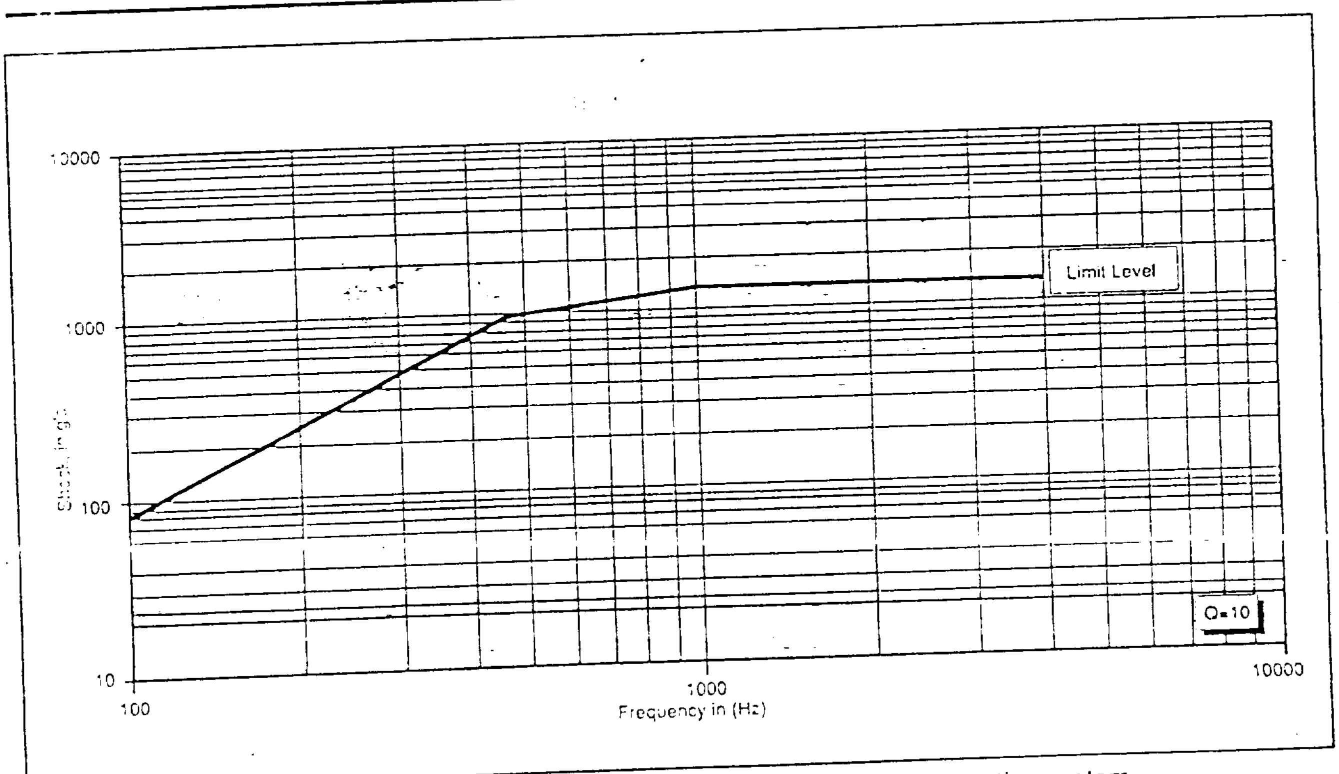


TABLE 7.6 . *Shock response spectrum constraint upon customer supplied separation system.*

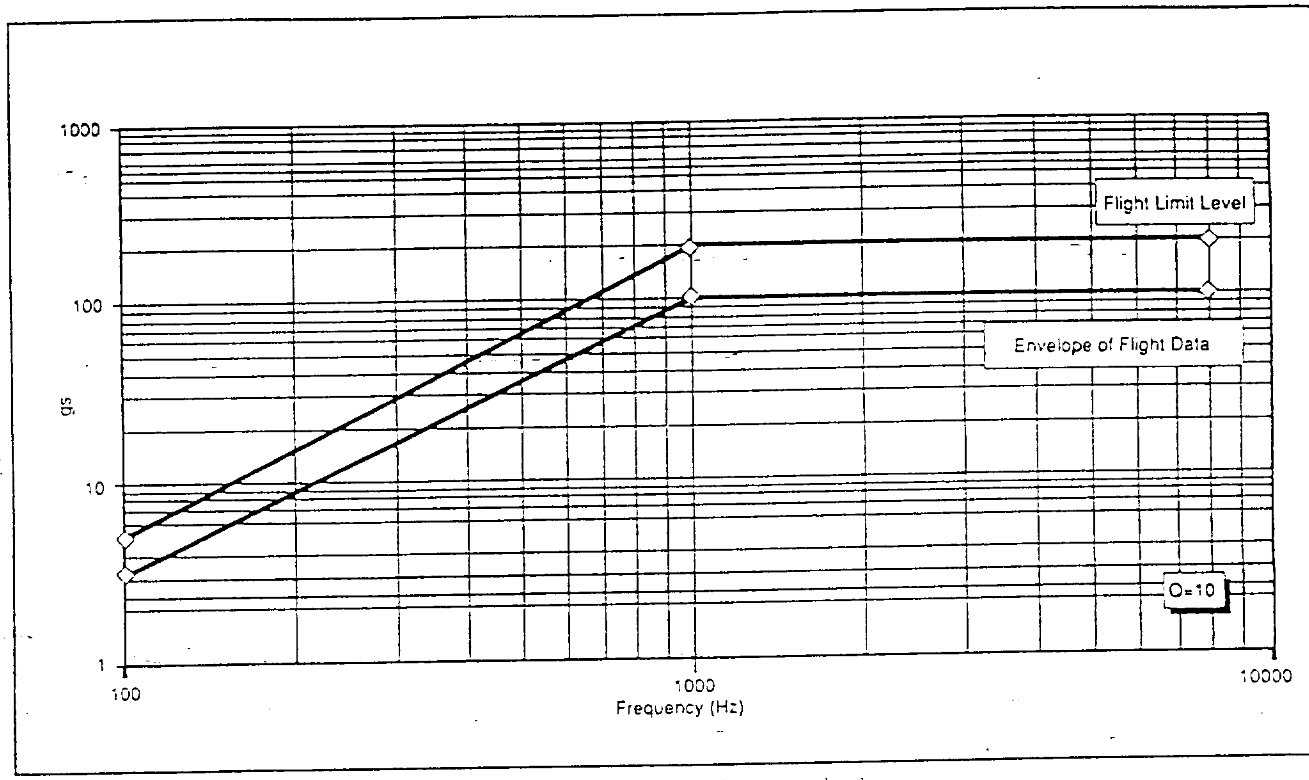


TABLE 7.7 . Payload shock environment (excluding payload separation).

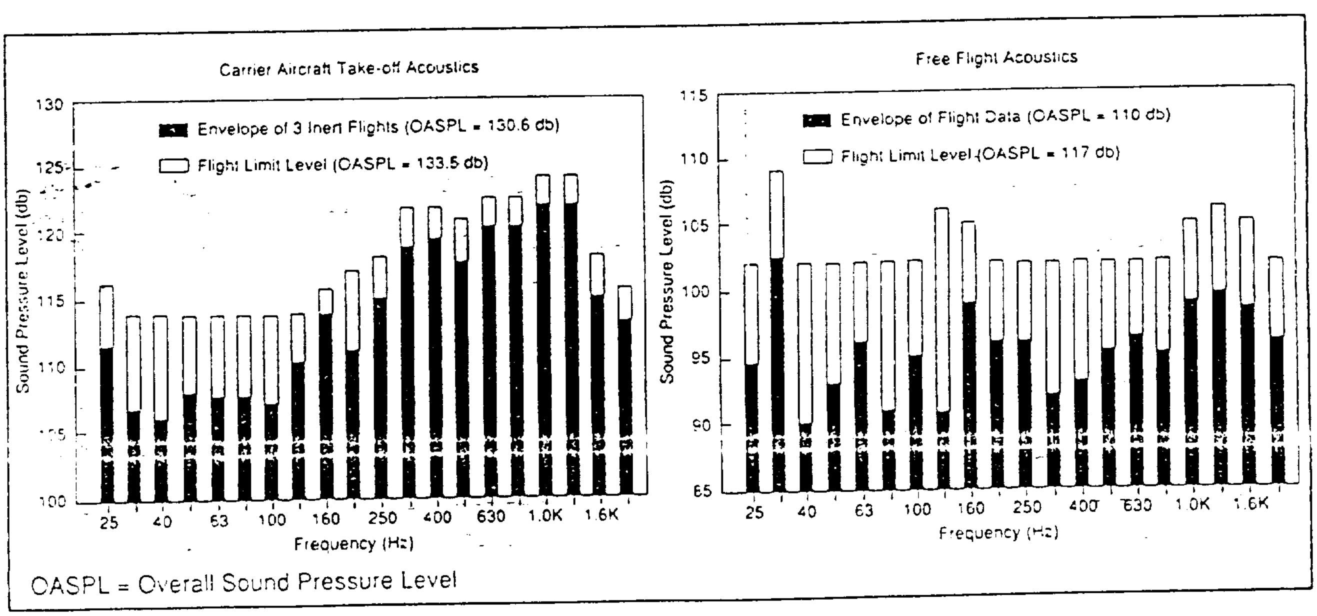


TABLE 7.8 . Payload acoustic environment.

8.0 THERMAL SUBSYSTEM

RINGSAT's orbit and structural configuration define the parameters for the thermal component of the satellite. The thermal system is composed solely of passive techniques used for moderation of the temperature extremes that the satellite will experience. Special considerations have been made for the electronics, battery and fuel temperature limits.

8.1 REQUIREMENTS

The requirements that must be met for the thermal subsystem are as follows:

1. Eliminate unwanted energy due to the body-mounted solar cells.

2. Provide a moderate temperature environment for the critical electronic, battery and fuel components.

3. Limit the amount of power required for the heaters to maintain temperature environment by utilizing other thermal control techniques.

8.2 DESIGN

Initial design calculations revealed a satellite temperature fluctuation of 72 degrees from -52.1 to 19.6 degrees Celsius when in the design orbit (Table 8-1). Specific equipment temperature limits are listed in Table 8-2. From these two sources it was determined that passive control features would be all that is necessary to regulate the temperatures within the satellite.

8.3 THERMAL COMPONENTS DESCRIPTIONS

Radiator -

From the very general, basic initial calculations it was determined that a radiator of a minimum area of 0.512 m^2 is necessary to eject excess heat out to space. This radiator is placed in the center of each of the eight panels. This was done for two reasons; (1) to allow for excess energy from the solar cells to escape out of top and bottom of satellite, and (2) to enable equipment sensitive to extreme temperature to be placed near the radiator and not the solar cells.

Battery and Electronic Equipment -

Special thermal considerations are taken into account for the battery and the communications electronics. A battery needs to be maintained within 5 to 20 degrees Celsius. In order to do this it is placed in an insulation shroud made of Aluminized Teflon and insulated by multi-layered insulation (MLI). One side is secured to the inner platform of the

satellite using fiberglass spacers and Titanium bolts to decrease effects due to conduction. One side is against the side of the satellite where the radiator is, covered in silvered Teflon and is the radiator for the battery. Heaters and thermostats are attached to the battery to maintain the temperature above the minimum temperature limit of 5 degrees Celsius. The electronic components are encased in a similar manner with MLI, heaters and thermostats.

Liquid Nitrogen Fuel Tanks -

The only thermal consideration that needs to be taken into account with respect to the liquid Nitrogen is that it does not reach its critical freezing temperature. Since liquid Nitrogen does not freeze unless exposed to temperatures down around 0 degrees Kelvin, the moderate temperature extremes experienced by the satellite will not have any effect on the fuel.

Solar Cells -

The solar cells are designed to have an absorptivity/emissivity ratio of nearly 1. The solar cells chosen are silicon with a Boron back surface field and multi-layered anti-reflective coating. This allows the solar cells to have an emissivity of 0.85 and an absorptivity of 0.65. These values cause the satellite to experience worst case hot temperatures of 58.0 degrees Celsius and worst case cold temperatures of -46.9 degrees Celsius. (Tables 8-3 and 8-4) Due to the location of the internal equipment of the satellite and the cells and the added MLI behind the cells and platform the temperatures will have little effect on the highly sensitive equipment.

Structure -

In order for the energy from the solar cells to have a more direct exit into space the hexagonal-shaped structure is open at the end facing the Earth and partially open at the other end. The end facing away from the Earth will have the adapter ring on it thus the end will be partially covered. This area will be painted white so as to allow minimal energy to enter the satellite. On the inside of the ring, aside from the structural crossbeams (also painted white), the satellite will be open. These qualities along with the location of the radiator and the solar cells with create an environment amiable to the thermal design requirements.

Silvered Teflon -

This passive thermal control device is used as a highly effective emitter with a very low absorptivity. Therefore, the radiator area is covered with silvered Teflon. Thus, the battery and electronic equipment will have the protection of a low absorber and will be able to get rid of the excess energy due to the high emitter.

White Epoxy Paint -

White paint covers all of the satellite's faces exposed to space that are not already covered with solar cells or silvered Teflon. This economical and efficient passive thermal control is all that is needed to ensure that the satellite does not experience too extreme of temperatures at any time in its orbit.

TABLE 8-1: PRELIMINARY THERMAL PERFORMANCE ESTIMATES FOR THE RINGSAT SPACECRAFT

No.	Item	Symbol	Value	Units	Source	Comment
1	Surface area of satellite	A	3.64	m^2	calculated	Total area of satellite
2	Diameter of sphere	D	1.076	m	calculated	Same surface area
3	Max power dissipation	Qw	170	W	Pin(max)-Pout(min	
4	Min power dissipation	Qw	80	W	Pin(min)-Pout(max	
5	Altitude	H	881	km		
6	Radius of earth	Re	6378	km		
7	Angular radius of earth	rho	1.073	rad	Eqn 1	
8	Albedo correction	Ka	0.989	-	Eqn 2	
9	Max earth IR emission at surface	ql	258	W/m^2	Fig 11-14	Use for worst-case hot
10	Min earth IR emission at surface	ql	216	W/m^2	Fig 11-14	Use for worst-case cold
11	Direct solar flux	Gs	1363	W/m^2	Fig 11-14	Use max value
12	Albedo	a	40	%	Fig 11-14	Use max value
13	Emissivity	epsilon	0.8	-		assume white epoxy paint
14	Absorptivity	alpha	0.3	-		assume white paint degrades
15	Worst case hot temp	Tmax	19.569	deg C	Eqn 3	
16	Worst case cold temp	Tmin	-52.077	deg C	Eqn 4	
17	Upper temp limit	Tu	35	deg C	TABLE 11-40	Assume 5 deg C thermal margin
18	Lower temp limit	Tl	5	degC	TABLE 11-40	Assume 5 deg C thermal margin
19	Radiator Area (worst hot condition)	Ar	0.512	m^2	Eqn 5	Assume no solar heat input, max dissipation, Tmax
20	Radiator Temp (worst cold condition)	Tr	-30.680	deg C	Eqn 6	Assume no solar heat input, min dissipation, Ar
21	Heater power req to maintain radiator at lower limit	Qn	138.585	W	Eqn 7	Assume no solar heat input, Ar, Tl

TABLE 8-2: TYPICAL TEMPERATURE RANGES FOR SELECTED SPACECRAFT COMPONENTS

Components	Typical Temperature Range, deg C
Electronics	0 to 40
Batteries	5 to 20
Solar Arrays	-100 to +100
Propellant, Hydrazine	7 to 35
Structures	-45 to +65
Infrared Detectors	-200 to -80

TABLE 8-3: EMISSION AND ABSORPTION PROPERTIES

Device	Absorptivity	Emissivity
Silicon solar cells	0.65	0.85
White epoxy paint	0.3	0.8
Silvered Teflon	0.08	0.66

TABLE 8-4: MAXIMUM AND MINIMUM TEMPERATURES

Component	Surface Area(m^2)	Max T (deg C	Min T (deg C)
Solar cells	2.318	58	-46.9
Reflector	0.84	34.8	-14.8

9.0 ELECTRICAL POWER SUBSYSTEM

9.1 REQUIREMENTS

The requirements to be met by the RINGSAT electrical power subsystem are to:

1. Supply a continuous source of electrical power to spacecraft loads during the mission life

2. Support power requirements for average and peak electrical loads

3. Ensure energy storage source is capable of operating during five year mission life within the environment of space

4. Ensure the energy storage source operates at the bus voltage and is fully charged during daylight

5. Create decentralized power distribution system to turn power on and off to the spacecraft loads

6. Provide power regulation and control to the electrical power system

9.2 POWER SOURCE

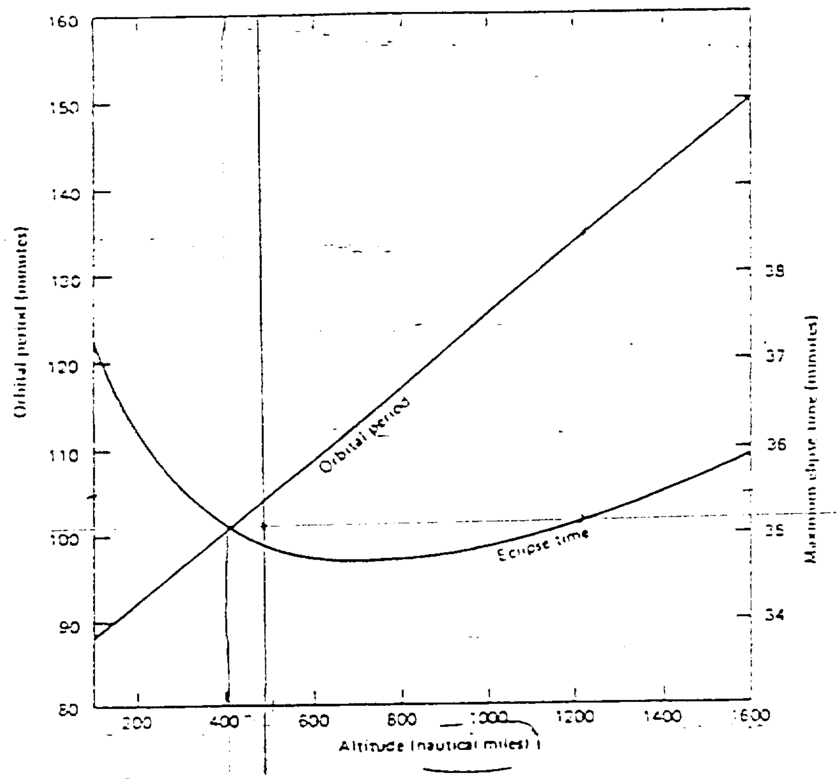
Photovoltaic solar cells convert incident solar radiation directly to electrical energy. The photovoltaic solar cell was chosen as the power source on RINGSAT versus using static power sources which use heat source for direct thermal-to-electric conversion or using dynamic power sources which use a heat source to produce electrical power using thermodynamic cycles.

The load profile of a spacecraft is a key determining factor in the design specifications of a power distribution subsystem. RINGSAT will use a 28 V dc power bus voltage. This bus voltage has been found to the best bus voltage according to Larson and Wertz. The 28 v bus works with with low power (< 2000).

9.2.1 SILICON SOLAR CELL

RINGSAT will use the Silicon K6700B WRAPTHRU Solar cells manufactured by Spectrolab, inc.. See APPENDIX 9.1 for specifications of the solar cell. The cell was chosen primarily due to the low solar absorptance and emittance. This reduced the amount of power that the thermal subsystem required.
The solar cells are body-mounted to the spacecraft. The body mounted configuration has an advantage in weight to deployed solar arrays due to the small power load requirement of the spacecraft.
The amount of time the spacecraft will spend in eclipse is 35.03663 minutes. Figure 9.1 below shows this graphically.

FIGURE 9.1



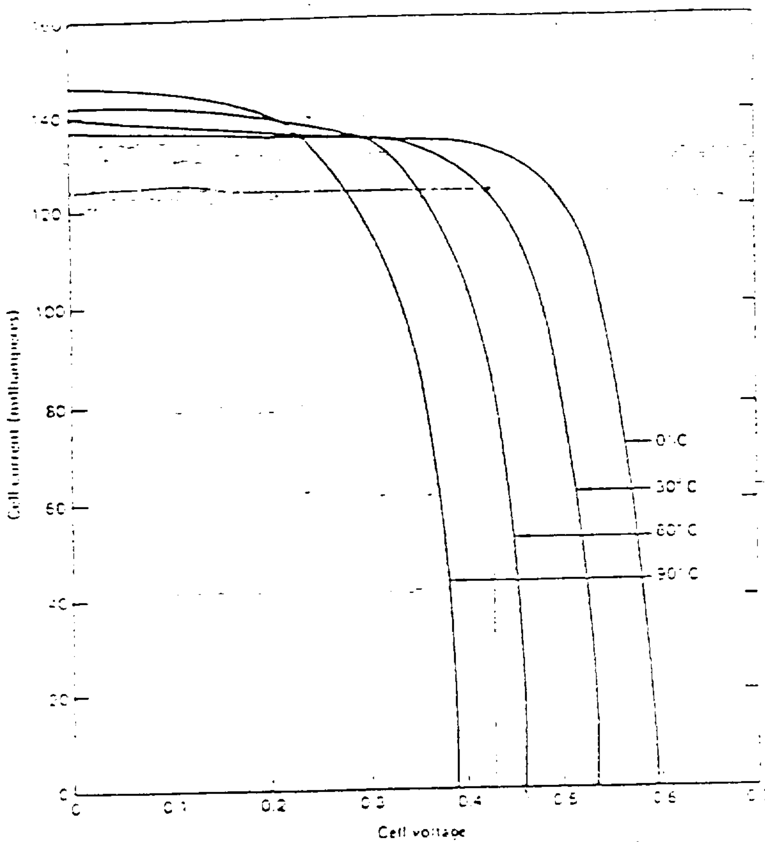
The maximum BOL cell current and maximum BOL power is determined from FIGURE 9.2 for a solar cell. The maximum power is 47.52 mW, the open circuit voltage is 0.46 volts, and the closed circuit current is 142 mA. These BOL values are used to define the body mounted array and depend on temperature.

The total number of cells in series is 62. The total number of strings is 17. Therefore, the total cells are 1054. These calculations are per side so that no matter what the orientaion of the body mounted cells, one face will be able to supply 170 Watts of power to the loads. The total area of the cells are 2.24 m^2. This will leave .07 m^2 total for wiring and packing. The total bare weight of the cell with no packing is 0.154 kg.

See APPENDIX for detailed calculations.

9.2.2 BACK SURFACE REFLECTOR

Back Surface Reflector cells are provided with a highly reflective metal surface between the solar cell wafer back surface and the cell's back side contact. Upon reflection of solar radioation having wavelengths greater than those absorbed by the cell (about 150 nm in silicon), this energy emanates from the cell into space. The solar absorptance is reduced significantly.



9.2.3 BACK SURFACE FIELD

Back surface field has an additional p^+ layer immediatly adjacent to the cells back surface. This field aids in the separation of photon generated electron-hole pairs and the collection of the minority carriers. From the JPL Handbook, a back surface field is more beneficial at the 1MeV fluence than an other type of silicon cell which include conventional, shallow junction, and textured. The back surface field on the silicon cell used on RINGSAT is a field of Boron. The atomic weight of Boron is 10.811 while the atomic eight of Aluminum is 26.9811. Also both Boron and Aluminum have 3 electrons in their outer shells.

TABLE 9.1
1 MeV Fluence Data from JPL Handbook
with 1.83 x 10^{13} Fluence and Back Surface Field

Short Circuit Current (mA/cm^2)	37.8
Open Circuit Voltage (Volts)	0.56
Maximum Power (mW/cm^2)	15.8
Voltage at Max Power (Volts)	0.458
Current at Max Power (mA/cm^2)	34.8

9.2.4 SHUNT DIODES

If a cell in series becomes shadowed, the amount of current limiting is reduced. Shunt diodes

are connected across shadowed cells. This produces a very low breakdown voltage. The diodes are connected acroos the submodules such that the shunt diode goes into forward conduction when the submodule is subjected to reverse bias.

9.2.4 ASSEMBLY DEGRADATION FACTORS

FIGURE 9.3
ELEMENTS OF INHERENT SOLAR ARRAY DEGRADATION

Elements of Inherent Degradation	Nominal	Range
Design and Assembly	0.85	0.77-0.90
Temperature of array	0.85	0.80-0.98
Shadowing of cells	1.00	0.80-1.00
Inherent Degradation, I_d	0.77	0.49-0.88

Table 11-35 Larson and Wertz

9.3 ENERGY STORAGE

A Nickel-Cadmium secondary battery will be used for the spacecraft energy storage. The battery converts chemical energy into electrical energy during dicharge and electrical energy into chemical energy during charge. The NiCd battery is alos necessary for pek power demands specifically when the spacecraft thrusters must fire to obtain the corect orientaion and when the gravity gradient boom is deployed.

9.3.1 TYPES OF ENERGY STORAGE

SECONDARY BATTERIES

The NiCd battery was chosen as the primary energy storage versus other secondary battery types primarily due to the maturity of the NiCd technology and the low cost of the NiCd.

The advantages of the NiCd include:

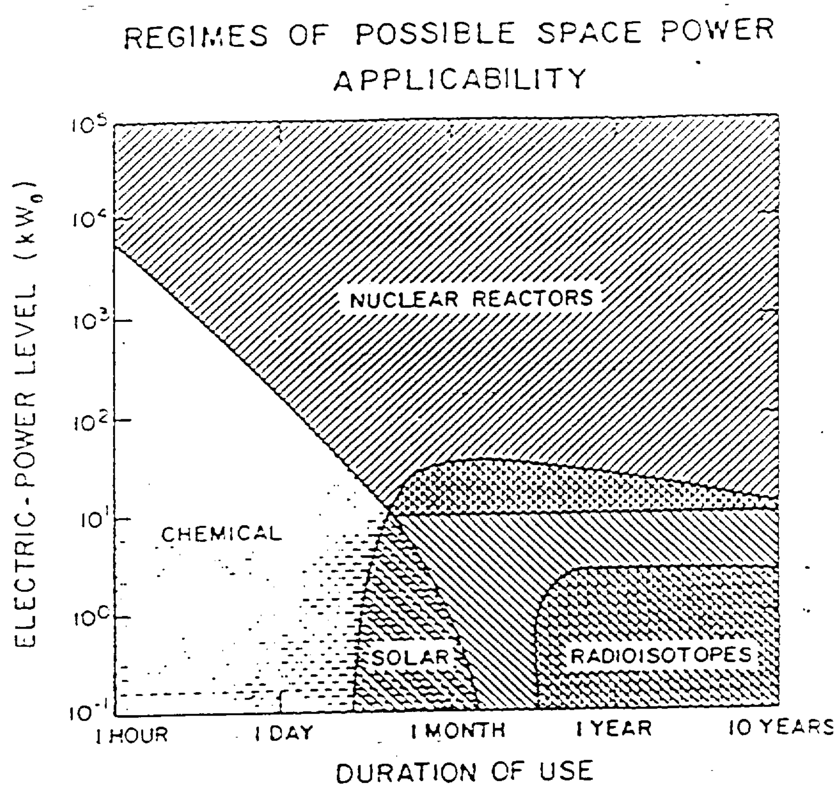
1. Most common type today
2. Low energy density (15-30 W-hr/kg)
3. Long cycle life
4. Good deep discharge tolerance
5. Can be reconditioned to extend life
6. Clean, rugged, lightweight

Silver-zinc (Ag-Zn), Nickel-hydrogen (NiH_2), and Lithium batteries also have their particular advantages; however, they do not fit into the low cost requirement of RINGSAT. NiH2 battery is 4 to 10 times more expensive than NiCd battery. This is the only advantage the NiCd has over NiH2, but cost is a driving requiremnt on the spacecraft.

FUEL CELLS
Fuel cells have the diadvantage of a fairly short duration and must carry a fuel and oxidizer. With a mission life of five years, the fuel cell would be inoperable by that time. The extra fuel and oxidizer would add weight to the spacecraft, therefore adding cost.

RTGS
Though an RTG (Radioisotope Thermoelectric Generator) has a long operating life, low to moderate weight, stored energy, and degradation due to particle irradition that is negligible, an RTG is costly. An RTG provides over 1000 Watts of power. RINGSAT only requires 160 Watts of power. An RTG would be overkill.

FIGURE 9.4



9.3.2 BATTERY TYPE

The Rated nameplate Capacity of the battery will be 14.25 A-hr [See Appendix]. The specifications for off-the-shelf NiCd batteries to obtain the capacity is a:

cell type model	VR10SF
size	SF
maximum bare height	0.089 meters
cell diameter	0.04115 meters
bare cell weight	0.390 kg

From the figure below, the discharge voltage(volts) for a 0.2C discharged capacity is 1.3 Volts.

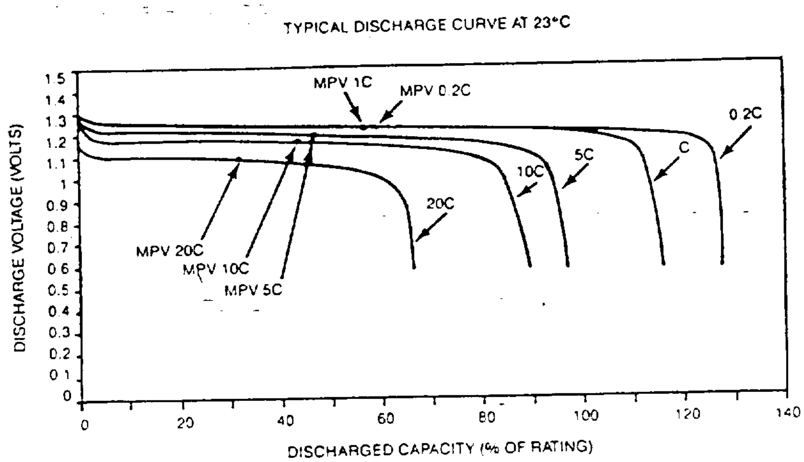


FIGURE 9. Typical discharge curves at 23C

In order to obtain a 28 volt dc bus voltage, 22 NiCd size SF cells will be placed in series.

The actual total mass is **9.009** kg.

The volume of the battery is **3.2227** cm^3.

The dimensions of the battery after packing are shown below. They will fit in the vented reqion of the spacecraft so as to stay as cool as possible. The optimal temperature range for the battery to operate is from 10 C degrees to 30 C degrees. The battery is able to operate in a range of -40 C to 45 C.

The battery will go through 25, 635.85 discgarge - charge cycles in the mission life of the spacecraft.

Assuming that the cost is $1000 per kW-hr for a sealed NiCad battery, the cost of the battery will be $498.75.

$$17.81\ A\text{-}hr * 28\ V\ bus\ voltage = 498.70\ W\text{-}hr = .49870\ kW\text{-}hr$$

$$0.49870\ kW\text{-}hr * \frac{\$1000}{kW\text{-}hr} = \$498.70$$

6.4 POWER DISTRIBUTION

The power distribution system on RINGSAT will be a decentralized system. The decentralized system regulates all the spacecraft loads within the elctrical power susystem.

An unregulated bus will be use din conjunction with the decentralized system and converters regulate power within the electrical power subsystem. Spacecraft power generates direct current. RINGSAT will use a 28 +/- 6 Volt dc bus voltage. In an unregulated subsystem, th eload bus voltage is the voltage of the batteries.

6.5 POWER REGULATION AND CONTROL

RINGSAT will use a Direct-Energy-Transfer (DET) power subsystem to control electrical power that is generated by the array to prevent battery overcharging and undesired spacecraft heating. For a DET system, a shunt regulator operates in parallel to the solar cells and shunts the current away from the subsystem when the loads or battery charging do not need power. A DET is very efficient, has fewer parts, lower mass , and higher effieciency at EOL than the Peak Power Tracker (PPT) power subsytem.

FIGURE 9.5

APPENDIX 1

SOLAR ARRAY DESIGN PROCESS

1. Determine requirements and constraints for power subsystem solar array design.

    *a. average power required during daylight and eclipse.

    P = 170 W

    (1) size photovoltaic system to meet power requirments at EOL

    b. Orbit altitude and eclipse duration

    (1) 475 nm = 881 km

    (2) T_e = 35. 03663 min (calculated)

    = 35.2 min (graph figure 3-19)

    *c. Design lifetime

    (1) 5 YEARS

2. Calculate the amount of power that must be produced by the solar arrays P_{sa} (power solar arry must provide during the daylight to power spacecraft for an entire orbit

$$\cos\frac{\Phi}{2} = \frac{\cos\rho}{\cos\beta_s}$$

$\frac{\Phi}{2}$ = *half of the rotation ∠ at maximum eclipse*

ρ = *angular radius of earth*

β_s = *∠ of sun above orbit plane*

$$T_E = P\left(\frac{\Phi}{360°}\right)$$

P = *orbit period*

T_E = *duration of eclipse circular orbit*

a. B_s = assumed inclination + angle between ecliptic and Earth's equator. = 83 deg

b. Ecliptic is defined as the path of the sun over the year- (23 deg, 26'21.448" (pg. 808 Larson and Wertz))

c. $P_e = P_d$ = 170 W

d. T_e = 35.2 min

T_d = 67.38 min

e. X_e = 0.65

X_d = 0.85

f. P_{sa} = 336.63 W

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d}$$

P_{sa} = *power provided during daylight*
P_e = *power requirements during eclipse*
P_d = *power requirements during daylight*
T_e = *length of eclipse per orbit*
T_d = *length of daylight per orbit*
X_e = *efficiency of power regulation during eclipse*
X_d = *efficiency of power regulation during daylight*

3. Select type of solar cell and estimate power output, P_o, with the sun normal to the surface of the cells

a. SILICON

b. P_o = 190 W/m^2 if the incident solar radiation (1358 W/m^2) is normal to the surface

c. ideal efficiency for silicon cell is 14.2%.

4. Determine the BOL power production capability, P_{BOL}, per unit area of the array.

$$P_{BOL} = P_o I_d \cos\theta$$

P_{BOL} = *power per unit area at beginning of life*
P_o = *solar cell power output*
I_d = *inherent degradation*
$\cos\theta$ = *cosine loss*
θ = *sun incidence* $\angle$

a. Inherent degradation includes design inefficiencies, shadowing and temperature variations. Ranges between 0.49 - 0.88 with a nominal value of 0.77.

b. The angle theta is measured beteen the vector normal to the surface array and the sun line. If theta is 90 degrees, P_{BOL} = 146 W/m^2.

5. Determine the EOL power production capapbility, P_{EOL}, for the solar array

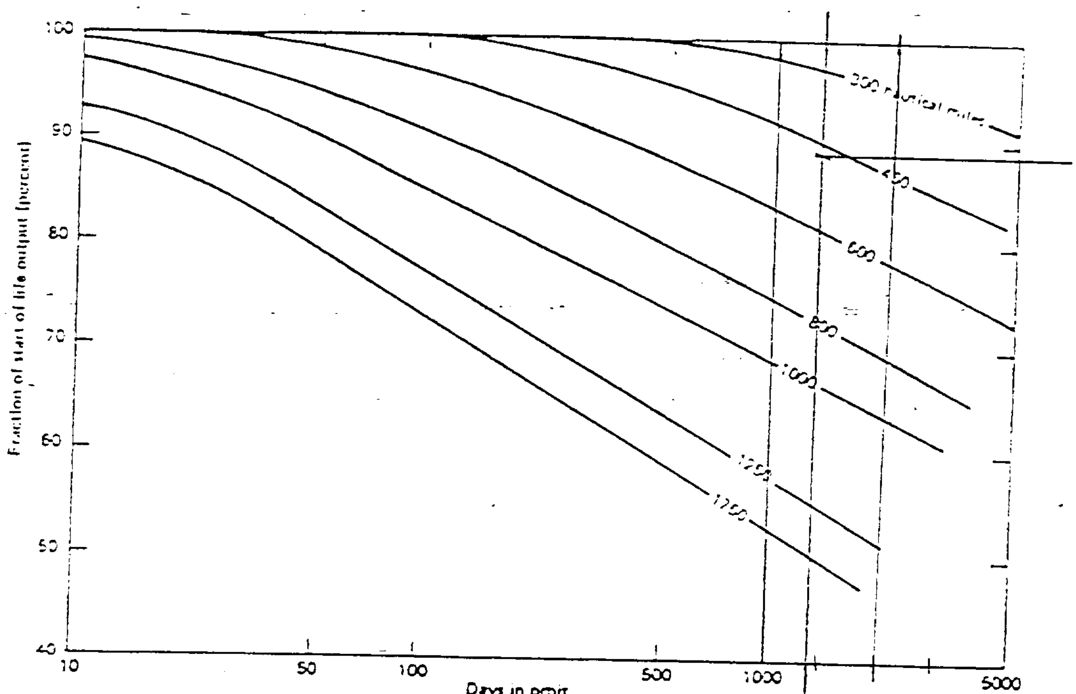
a. assume degradation per year = 3.75 % for LEO silicon cell
b. The life degrdation is 0.83. See FIGURE 9...

$$L_d = \left(1 - \frac{degradation}{year}\right)^{satellite\ life}$$

L_d = Life degradation
satellite life = 5 years

c. $P_{EOL} = 120.6\ W/m^2$

FIGURE 9.
POWER DEGRADATION AS A FUNCTION OF TIME
ORBIT INCLINATION 60



6. Estimate the solar cell area, A_{sa}, required to produce the necessary power, P_{sa}, based on P_{EOL}.

a. two ways to calculate area

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$

$$A_{sa} = \frac{P_{sa}}{\eta\ 1358}$$

b. $A_{sa} = 2.79\ m^2$

APPENDIX 2

Solar Cell Design

* Data Required from JPL Handbook
  - $I_{max\ power}$ = 34.8 mA/cm^2 (fig 3.75)
  - $V_{max\ power}$ = 0.458 V (fig 3.74)
  - P_{max} = 15.8 mW/cm^2 (fig 3.73)
  - density of silicon = 5 kg/m^2
  - utilization factor = 0.85

* Assumptions
  - cell thickness = 200 microns
  - coverslip thickness = 0.335 g/cm^2
  - Bus Voltage = 28 V
  - Load = 170 W
  - MeV Electron Fluence = 1.85 x 10^{13}

* Number of Cells in series

$$Series\ cells = \frac{Bus\ voltage * 1.05}{V_{mp}}$$

$$Bus\ voltage = 28\ V$$
$$V_{mp} = 0.472\ V$$
$$\therefore Series\ cells = 62$$

1.05 takes into account any line losses

* Number of Cells in Parallel

$$\|\ strings = \frac{Total\ cells}{Series\ cells}$$

$$Total\ cells = 1037\ cells$$
$$Series\ cells = 62\ cells$$
$$\therefore \|\ strings = 17\ strings$$

* Total number of cells

$$Total\ cells = \frac{Power\ needed}{Power\ per\ cell}$$

$$Power\ needed = 170\ W$$
$$Power\ per\ cell = 0.164$$
$$\therefore Total\ cells = 1037$$

* Dimensions of solar cell
  1.33 cm X 2.0 cm

* Area per cell
  2.66 cm^2 = 0.000266 m^2

* Total Area

$$Total\ area = Total\ cells * Cell\ area$$

Total Cells = 1054
Total Area per side = 0.28 m^2
Total Area (8 sides) = 2.24 m^2

* Total area taking into account wiring (utilization factor)

$$Panel\ area = \frac{Total\ area}{Utilization\ Factor}$$

Area = 2.63 m^2

APPENDIX 3

Rated Capacity of NiCd Battery

* Current

$$current = \frac{Load\ during\ darkness}{dc\ bus\ voltage}$$

Load during darkness = 170 *W*
dc battery voltage = 28 *V*
current = 6.0714 *A*

* A - hr

$$A\text{-}hr = current * eclipse\ time$$

current = 6.0714 *A*
eclipse time = 0.5867 *hr*
Ahr = 3.562 *A-hr*

* Depth of Discharge: 20 %

* Nameplate

$$Nameplate = \frac{A\text{-}hr}{DOD}$$

A-hr = 3.562 *A-hr*
DOD = 0.20
Nameplate = 17.81 *A-hr*

APPENDIX 4

Required Capacity of Secondary Battery

* NiCd energy density = 25 - 30 W hr/kg

* Stored Energy needed

$$P * T_E = \textit{Stored Energy Needed}$$

$P = 170\ W$
$T_E = 0.5867\ hr$
$\therefore$
Stored energy needed = 99.739 W-hr

* Depth of Discharge: 20 %
  see fig 11-11 Larson and Wertz

* Battery Capacity

$$\textit{Battery Capacity} = \frac{\textit{Stored energy needed}}{DOD}$$

Stored energy needed = 99.739 W-hr
DOD = Depth of Discharge = 20 %
$\therefore$
Battery capacity = 498.695 W-hr

* Mass

$$M = \frac{\textit{Battery capacity}}{\textit{Energy density}}$$

Battery capacity = 498.695 W-hr
Enrgy density = 30 $\frac{W\text{-}hr}{kg}$
$\therefore$
Mass = 16.623 kg

* Packing factor
  assume a packing factor of 5 % of total mass(very safe assumption)

* Total Mass

$$Total\ M = PF * M$$

PF = Packing factor = 0.05
M = 16.623 kg
$\therefore$
$TotalMass$ = 17.45 kg

SILICON
K6700B
WRAPTHRU SOLAR CELLS

SPECTRODATA

TYPICAL ELECTRICAL PARAMETERS*

Jsc = 41.9 Milliamperes/cm^2
Jmp = 38.4 Milliamperes/cm^2
Vmp = 0.500 Volts
Pmp = 19.2 Milliwatts/cm^2
Voc = 0.618 Volts
Cff = 0.74
Efficiency 14.2% Minimum Average
***AMO Sunlight (135.3 mw/cm^2), 28°C**

RADIATION DEGRADATION*

PARAMETER	1x10^{13}	1x10^{14}	5x10^{14}
Isc/Isc$_0$	0.99	0.94	0.85
Imp/Imp$_0$	0.99	0.95	0.85
Vmp/Vmp$_0$	0.95	0.88	0.82
Voc/Voc$_0$	0.96	0.89	0.84
Pmp/Pmp$_0$	0.94	0.84	0.70

***Fluence e/cm^2 1 MeV Electrons**

THERMAL PROPERTIES

Solar Absorptance = 0.65 (CMX)
Solar Absorptance = 0.63 (Fused Silica)
Emittance (Normal) = 0.85 (CMX)
Emittance (Normal) = 0.81 (Fused Silica)

WEIGHT

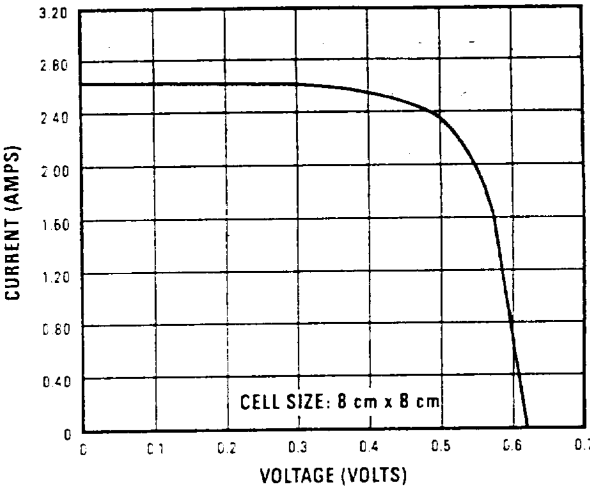
55 Milligrams/cm^2 (Bare)

TEMPERATURE COEFFICIENTS

Isc = +20.0 Microamperes/cm^2
Vmp = −2.15 Millivolts/°C
Voc = −1.96 Millivolts/°C

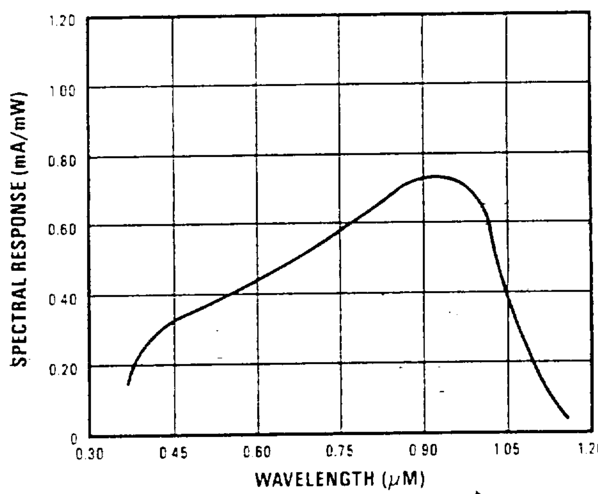
APPLICATION NOTES

TYPICAL I-V CHARACTERISTIC CURVE*



***AMO Sunlight (135.3 mw/cm^2), 28°C**

SPECTRAL RESPONSE



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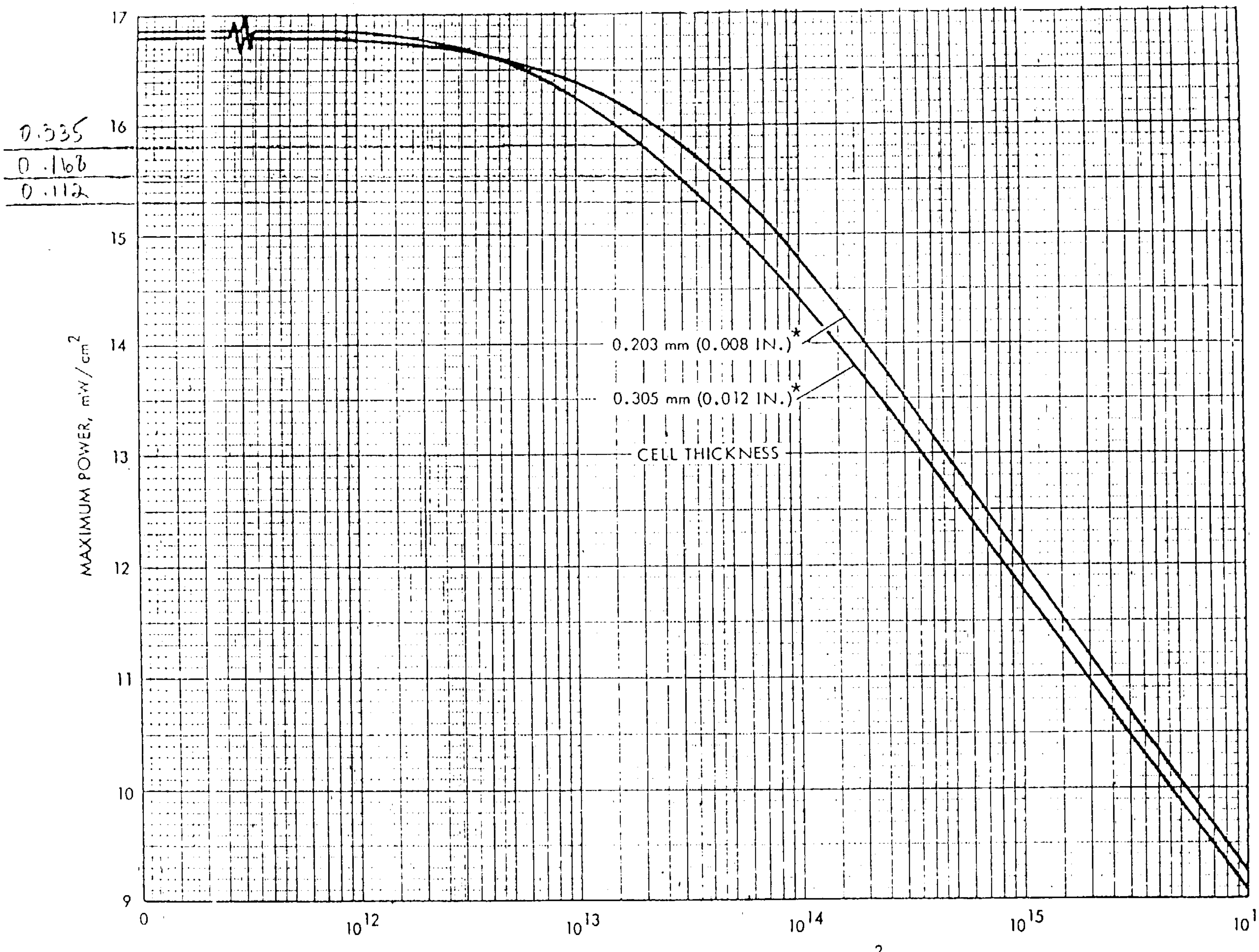

Figure 3.73 Maximum Power vs 1 MeV Electron Fluence for 10 Ohm-cm n/p Back Surface Field Silicon Cells. At 135.3 mW/cm² AMO Illumination, 30°C

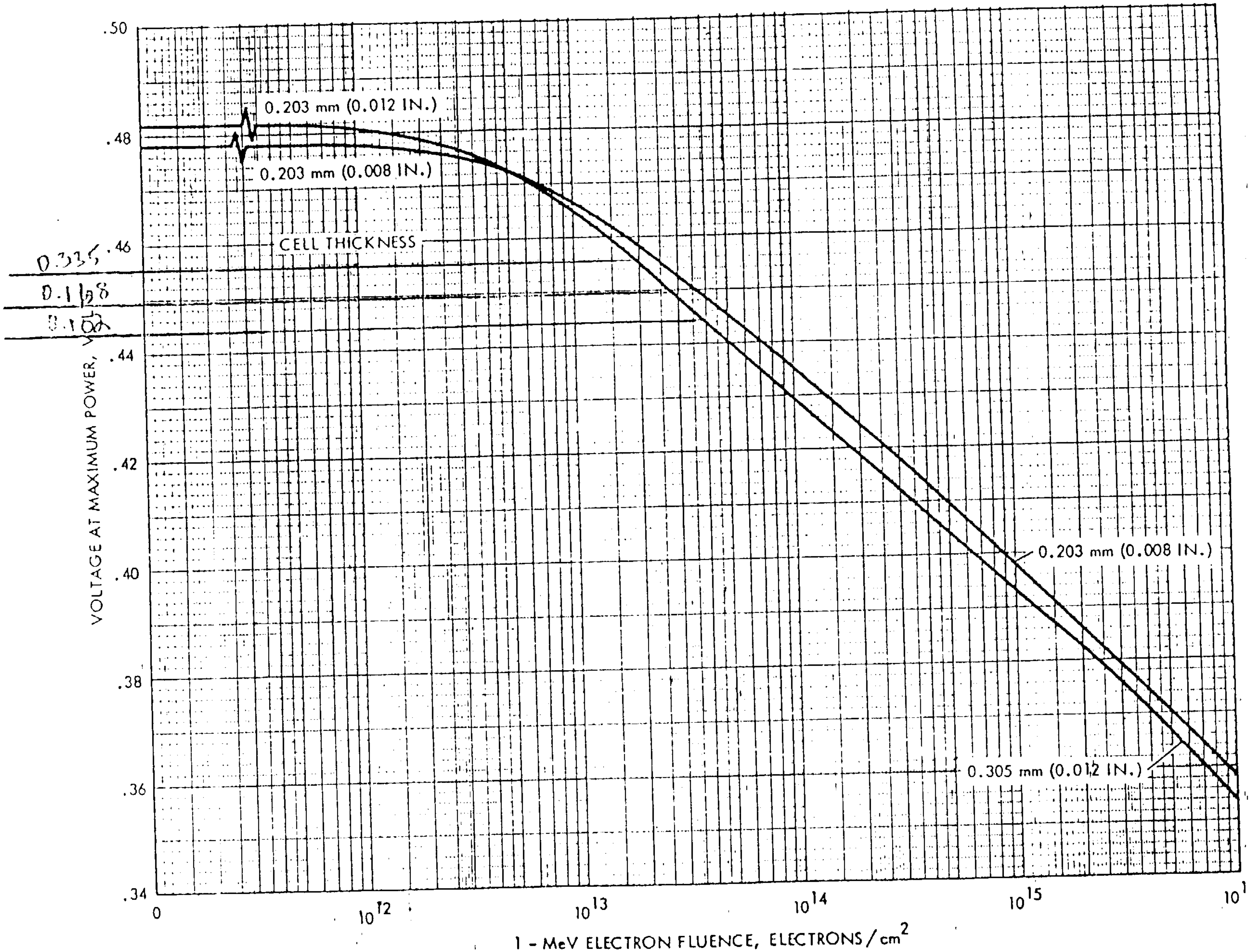

Figure 3.74 Voltage at Maximum Power vs 1 MeV Electron Fluence for 10 Ohm-cm n/p Back Surface Field Silicon Cells.
At 135.3 mW/cm^2 AMO Illumination, 30°C

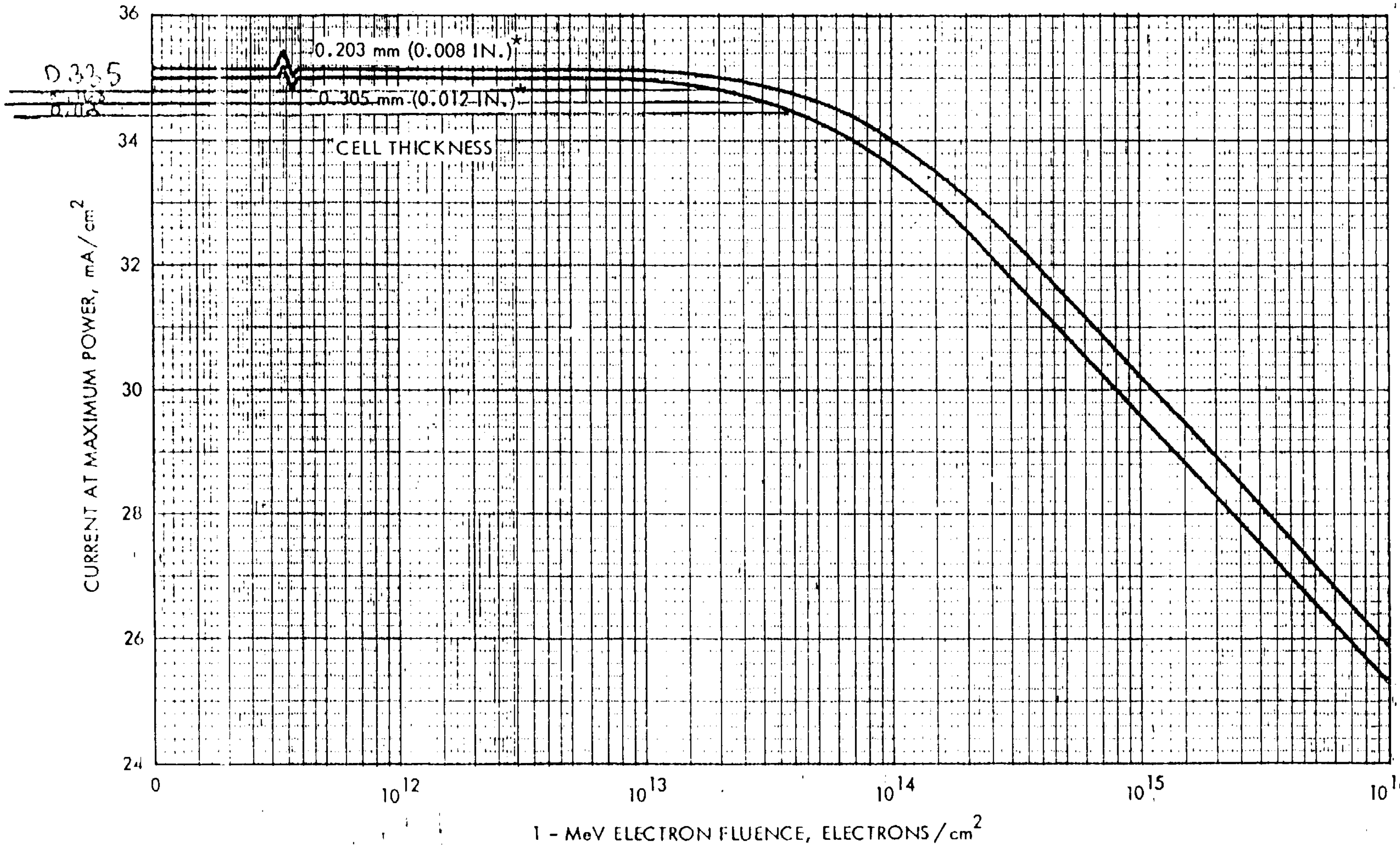

Figure 3.75 Current at Maximum Power vs 1 MeV Electron Fluence for 10 Ohm-cm n/p Back Surface Field Silicon Cells.
At 135.3 mW/cm^2 AMO Illumination, 30°C

NICAD®

SAFT manufactures a full line of NICAD batteries from a ⅓ AAA cell rated at .070 ampere hours to a Super F cell rated at 10.0 ampere hours. Our complete range of batteries includes standard cells, high temperature Polytemp® cells and fast charge, low resistance VR cells. Our VE series, available in AA, C and D sizes, is designed for rapid charge, high energy applications such as cordless appliances. SAFT button cells are reliable, high quality units offering unmatched performance and extremely long [illegible] life. Our new [illegible] batteries [illegible] ultra-thin to provide [illegible] usable space for today's small personal appliances.

SAFT has a cell available to solve even the most difficult battery requirements. Additional information on any of these products is available through the offices listed on the back cover.

NICAD® STANDARD CELLS

Cell Type Model	Size	Max. Bare Height in.	Max. Bare Height mm	Cell Diameter in.	Cell Diameter mm	Capacity AH Rated @ 20°C 1C	Capacity AH Rated @ 20°C 0.2C	Charge Rates @ 20°C Continuous amps	Charge Rates @ 20°C Continuous hrs.	Charge Rates @ 20°C Maximum amps	Charge Rates @ 20°C Maximum hrs.	Bare Cell Weight oz.	Bare Cell Weight grams	Int. Resist mΩ	Vent Type H=Hermetic R=Resealable
.070SC	⅓AAA	.825	20.96	.376	9.55	.060	.070	.007	16	.023	[illegible]	.14	4.0	220	[illegible]
.110RC*	⅓AA	.660	16.76	.535	13.59	.100	.110	.011	16	.037	4.0	.23	6.5	80	R
.150SC	⅓A	.635	16.13	.593	15.06	.140	.150	.015	16	.030	8.0	.28	7.9	100	[illegible]
.180RC	AAA	1.760	44.50	.410	10.50	.160	.180	.018	16	.036	8.0	.35	10.0	120	R
.225RC	⅓AF	.665	16.89	.652	16.56	.200	.225	.023	16	.045	8.0	.38	10.8	30	[illegible]
.250RC*	½AA	1.192	30.28	.535	13.59	.225	.250	.025	16	.080	4.0	.38	10.8	60	R
.450RC	⅔AF	1.100	27.94	.652	16.56	.405	.450	.045	16	.150	[illegible]	.55	15.6	48	[illegible]
.500RC*	AA	1.967	49.96	.535	13.59	.450	.500	.050	16	.165	4.0	.70	19.9	22	R
.600SC	A	1.920	48.77	.593	15.06	.530	.600	.060	16	.120	[illegible]	.85	24.0	25	[illegible]
.750RC	½C	.925	23.50	1.012	25.70	.660	.750	.075	16	.150	8.0	1.10	31.0	36	H
.850RC	AF	[illegible]	49.28	.652	16.56	.720	.850	.080	16	.265	[illegible]	1.05	30.0	[illegible]	[illegible]
1.0RC	⅗C	1.150	29.21	1.012	25.70	.880	1.000	.100	16	.200	8.0	1.40	39.7	18	H
1.2RC	Cs	1.652	41.96	.870	22.25	1.060	1.200	.120	16	.240	8.0	1.60	45.0	16	[illegible]
1.8RC	CsD	2.225	56.52	.876	22.25	1.800	2.000	.180	16	.367	8.0	2.40	68.0	8	R
2.0RC	C	[illegible]	[illegible]	[illegible]	25.70	1.800	2.000	.200	16	.400	8.0	2.40	68.0	[illegible]	[illegible]
2.2RC	½D	1.448	36.78	1.275	32.39	2.000	2.200	.220	16	.440	8.0	2.80	80.0	10	R
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
7.0RC	F	3.508	89.10	1.275	32.39	6.300	7.000	.700	16	1.400	8.0	7.40	210.0	7	R
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	16	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

[illegible]